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ROYAL AIRCRAFT ESTABLISHMENT

F A R N B O R O U G H, H A N T S

TECHNICAL NOTE No: I.A.P.1042

INTERIM NOTE ON THE
DEVELOPMENT OF AN AUTOMATIC PILOT FOR HELICOPTERS

by

G.R.COOPER, M.Sc., A.Inst.P. and

M.C.CURTIES, B.Sc., A.F.R.Ae.S.

SEPTEMBER, 1955

M I N I S T R Y O F S U P P L Y

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Corrigenda to Technical Note No. IAP. 1042

November, 1955.

ROYAL AIRCRAFT ESTABLISHMENT, FARNBOROUGH

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Fig.24

For $T = W \cos(\theta + \eta) \approx W$

read $T = W \sec(\theta + \eta) \approx W$

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U.D.C. No. 629.13.014.59 : 629.135.4

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SUMMARY

The effectiveness of the helicopter in Sonar dipping is limited by its poor flying qualities. The Note describes the development so far of auto-control equipment designed to improve the machine's suitability for this role and, in addition, to provide a manoeuvre-holding autopilot for cross-country flying. The system uses electric-spring rate-gyros with integration, magnetic amplifiers and d.c. electric servos; operation is through the pilot's normal controls. All the development to date has been done in a Whirlwind H.A.S. Mk.22, although the Service use of the equipment will be in the Naval helicopter N.A.43 and its R.A.F. equivalent; no machines of this type are yet available for autopilot work.

The equipment will eventually provide facilities for automatic hovering with Sonar, height control and probably coupling to navigation equipment when available.

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1 Introduction

The autopilot development to be described was begun mainly in response to a requirement by D.A.W., Admiralty, for an automatic hovering aid for Dipping Sonar use in the Naval helicopter N.A.43. Somewhat later the Air Staff issued O.R.942 for a helicopter autopilot with facilities for cross-country flying as well as hovering, and it was agreed that the autopilot work would take account of this requirement also.

During preliminary discussions with Admiralty representatives, including helicopter pilots, it became apparent that pilot fatigue was an important factor in limiting the accuracy and efficiency of Sonar dipping, and that a major cause of this was the inherent instability of the helicopter itself. It was agreed, therefore, that a stabilising device, even though not giving fully automatic hovering, might well be of great value, and that the development should be tackled in stages, a flight assessment being made at each stage to weigh the benefits gained against the cost and complexity of the equipment required. The stages of development were to be:

(i) Artificial stabilisation, the pilot exercising overall control by means of either a miniature stick or the main cyclic control stick, suitably adapted.

(ii) Automatic plan-position control over the Sonar, height being controlled by the pilot.

(iii) Automatic flight over the whole speed range, the helicopter being held to an established attitude and heading; automatic turn co-ordination not required.

(iv) Automatic height control, via collective pitch if necessary, e.g. in hovering.

It should be noted that whilst stage (i) is a primary essential, the other stages are largely independent of each other so the order of development listed is not technically necessary.

The present Note describes the progress made up to the end of February 1955. Briefly, stages (i) and (iii) have been completed; stage (ii) is under way; stage (iii) follows more easily and logically from stage (i) than any other, hence the inversion of order. All the flying to date has been done in a Whirlwind H.A.S. Mk.22 loaned to R.A.E. by the Navy; the equipment has been made mainly by Messrs. Louis Newmark Ltd. who have a development contract for this project and who are co-operating closely with I.A.P. Dept. R.A.E. on all aspects of the development and flight trials. Since, however, the machine for which the autopilot is intended is the Bristol 191, a twin-rotor helicopter, which will not be available for this work until the middle of 1956, further flight trials are to be carried out in a more representative machine, the Bristol 173. Part of this development, which is unlikely to begin before the end of 1955, is the integration of the autopilot and power-controls, a considerable task in itself; it will be seen therefore that a good deal of faith is implied in the validity of the results to date as applied to the Bristol 191, if the equipment is to be in production to meet this aircraft.

It should be mentioned, finally, that an assessment of stage (i) with the addition of a heading monitor, has already been made using the Sonar equipment in typical sorties, with encouraging results.

2 Description

The development equipment is known as the Autopilot Type K. Before describing the various units of the autopilot and their functions, it is proposed to stage what, in its present form, the equipment does. As a stabiliser (stage (i)) it confers on the aircraft "hands-off" stability, but does not make it hold any datum attitude or heading. The pilot, via his controller, maintains or modifies the flight condition as required. There is, however, a short-memory position term in the pitch and roll controls so that after a brief disturbance, the machine attempts to return approximately to the attitude it held immediately beforehand. Manoeuvres may be carried out by use of the pilot's controller, which produces an angular velocity of the machine about the appropriate axis proportional to controller deflection or force. As an autopilot (stage (iii)), it will attempt to hold the attitude and heading of the aircraft existing at the time of engagement; there are no facilities for holding a co-ordinated turn if engaged during one. If a turn is to be made under autopilot, a "Manoeuvre" button on the pilot's controller is pressed and the control system reverts to its stabiliser mode, when the pilot's controller may be used to make turns which are easily co-ordinated by the pilot. The present pilot's controller is integrated with the normal cyclic and yaw controls, by the use of force-sensitive signal pick-offs which bias the autopilot gyros.

The equipment may be engaged as a stabiliser in any attitude or trim condition either in flight or before take-off: before switching to autopilot operation, a few seconds in the desired steady flight condition are necessary.

A discussion is given in Appendix I of the stability of the helicopter; it may be summarised here by remarking that the helicopter is usually dynamically unstable in pitch and roll both in hovering and cruising flight, and that in order to stabilise it, control must be applied proportional to the angular displacements and velocities of the fuselage. In the autopilot, signals embodying these quantities are obtained from electric-spring rate gyroscopes fixed to the aircraft, their circuits being arranged to give "short memory" integration and produce the required combination of signals directly.

This "short-memory" integration, produced by a leaky condenser in the gyro electric-spring circuit, means that a rapid change in aircraft attitude will produce, in addition to the rate signal during motion, a signal which, immediately after the change has occurred, is proportional to the angular displacement, but which will slowly die away. Provided the rate of decay is sufficiently small compared with the frequency of the helicopter's controlled response, this signal is effectively a position signal, but does not entail an angular datum which, in manoeuvring, the pilot does not desire.

3 Autopilot Units

The autopilot in the Whirlwind comprises eleven units, viz:

3-axis Gyro Unit

Amplifier

2 Servomotors for cyclic pitch control
(Linear actuators with 2" stroke)

1 Servomotor, Type 4B for yaw control

Yaw Compensator, coupled to the collective
pitch control

Switch Unit

Vertical Reference

Heading Reference

Pilot's controller, incorporated with the normal aircraft controls

Inverter

In addition an emergency release trigger is fitted on the collective pitch lever and a manoeuvre button is provided on the top of the cyclic stick. Photographs of the units in their installed positions are given in Figs.8 to 14.

In the design, an effort has been made to use as many standard components as possible, so as to save development time and to ease the servicing and spares requirements on the final equipment.

3.1 Gyro Unit (Fig.18)

This contains three electric-spring rate-gyros essentially similar to those used in the Type B Autopilot[1] and the Mk.1 Autostabiliser. They are arranged to detect fuselage angular velocities in pitch, roll and yaw, and the electric-spring feedback circuit is completed in the Amplifier so as to obtain pure rate or combined rate and position signals as desired (Fig.1). Two additional torque coils on the gyros receive pilot's demand signals and, in the pitch and roll gyros, monitor signals from the vertical reference when required. These gyros have a threshold sensitivity of less than 2°/min but, by reason of spurious gymbal-bearing torques, may give varying signals of this order with no rate input. This effect, shown as a drift in aircraft attitude in the "Stabiliser" mode (see below) can only be offset by frequent use of trimmers and efforts are being made to eradicate it in the gyros themselves.

3.2 Amplifier (Fig.17)

The amplifier contains not only the three magnetic amplifiers and power transformers but also all the circuit elements for attenuating or operating on signals, the control relays for the servomotors and the relays required for circuit switching to select to various modes of operation.

The magnetic amplifiers, essentially the same as those developed for the Autostabiliser Mk.2, operate from a 10 volt, 400 c/s supply and give a push-pull output of about 1/8 watt. They each have five input windings, giving current gains of 24, 24, 12, 6, 6, and are designed to operate with loads of 20 ohms per side; they are here feeding the coils of Carpenter polarised relays Type 5HM 4A of 40 ohms, shunted by 47 ohm resistors.

Also in the Amplifier are the integrating condensers, 500 µF reversible electrolytic, in the electric-spring gyro circuits, shunted by resistors of several thousand ohms to give the short-memory position signal; phase-advance condensers, also reversible electrolytic, are fitted in the signal lines to improve short period stability.

The servomotor control circuits are as already developed either for the Autostabiliser or the Type B autopilot (Figs.4, 5).

In order to permit selection of the autopilot mode of operation in any flight condition, the signals of the vertical reference are continuously followed up by two small servo-units in the Amplifier, each comprising a Mess-motor, gear train and potentiometer. The monitor signals to the rate-gyros are the difference between reference and input, nominally zero until

the autopilot mode is selected, when the follow-up is locked and subsequent changes in attitude give rise to appropriate signals.

3.3 Servomotors

(a) Cyclic (Fig.19). The fact that the cyclic controls on the Whirlwind are provided with power boost has enabled small servomotors to be used as in the Autostabiliser Mk.1. They each comprise a 10-watt d.c. shunt motor driving an irreversible nut and leadscrew output via low-ratio spur gears; the output rod is coupled into the cyclic pitch controls at the pilot's end below the cockpit floor. The body of the servo is not fixed to the aircraft, but is carried on an arm linked to one member of an electromagnetic clutch whose other member is fixed to the aircraft; the whole servo can therefore move freely with the cyclic controls during manual flying whilst the clutch is disengaged, but will operate the cyclic controls when the electromagnetic clutch is engaged for autopilot flying. Override of the electromagnetic clutch in emergency is easy, requiring only about 15 lb-ft; but it can be disengaged immediately by the emergency release trigger on the collective pitch lever. Another advantage of this arrangement is that the aircraft may be trimmed prior to autopilot engagement with the servo at mid-stroke, so that no additional electric trimming signals are necessary, and the full servo stroke is available for control irrespective of the aerodynamic trim required. The servomotor shunt field is continuously energised, control being by relays supplying the armature reversibly; a velocity-feedback signal for local stability is obtained from the armature back-e.m.f., and a position feedback signal is derived from a potentiometer and pick-off on the lead-screw. Maximum servo effort is 30 lbs; the stroke is ±1", giving full cyclic control and the maximum control speed is 2" in $1\frac{1}{2}$ secs.

(b) Yaw (Fig.20). As there is no power boost in yaw on the Whirlwind a larger servo was required than for the cyclic controls and the R.A.E. Type 4B magnetic-powder clutch servo was adopted. This gives a rotary output of 40 lb ft maximum torque over an angular travel of ±30°; maximum speed of operation is 20°/sec on no load. The servo contains a motor driving in opposite directions the input shafts of two magnetic-powder clutches, whose outputs couple via a gear train to a common servo-output shaft. The clutches are operated by a polarised relay in a purely on-off fashion, but a local relay feedback circuit gives a quasi-continuous control effect by inducing a high-frequency oscillation. A small permanent magnet motor is geared up from the output shaft of the servo to give a velocity feedback signal for stabilising. A plate clutch, electrically operated, is included in the output of the servo to permit disengagement.

The Bristol 191 will be fitted with power controls into which the autopilot will feed signals via transducers. The servos used in the cyclic controls of the Whirlwind for these trials are not likely, therefore, to be perpetuated in any numbers, and little development effort has been spent on them. The Type 4B is in current use on other projects.

3.4 Yaw Compensator (Fig.10)

This comprises a simple potentiometer and pick-off, housed in a box fixed to the aircraft, the pick-off being coupled by an adjustable link to the collective pitch lever. A signal is thereby obtained in response to variations in collective pitch setting which is fed into the autopilot yaw channel to compensate approximately for the variations in rotor torque. Any residual yawing moments may be corrected by the autopilot heading signal at the expense of negligible heading errors; full compensation could be obtained by a yaw integral control, but only as a slow-acting element and this would be of little value during the hovering operation.

3.5 Switch Unit (Fig.21)

This contains the power switch for the autopilot, the channel selector switches in the clutch lines for engaging and disengaging the servomotor clutches to permit the use of channels independently, the selector switch for different modes of operation and the trimmers for off-setting rate-gyro drift.

The selector switch has four positions, viz. Standby, Stabilise, Cruise and Hover. In "Standby", all the autopilot circuits are energised, except the servomotor clutches; the servomotors centre themselves and the rate gyros give pure rate signals, the integrating condensers being short-circuited.

In "Stabilise", the clutches are engaged, the short memory integration is introduced in pitch and roll, the yaw signal remaining as a rate signal. The Messmotor units in the amplifier continuously follow up the vertical reference signals and the heading pick-off is unclutched from its gyro, so that these remain ineffective as regards control. The pilot's controller feeds demands as required to the rate gyros.

In "Cruise", the heading pick-off is clutched to the gyro and the vertical gyro follow-ups are fixed by switching the Mess-motor armatures to the short-circuit condition. The pilot's controller is switched out of circuit, but the trimmers remain effective.

In "Hover", the heading signal is retained so that the aircraft maintains direction, but the pitch and roll circuits revert to the "Stabilise" condition, the pilot exercising overall control about these axes via his controller.

3.6 Vertical and Heading References (Fig.14)

It is intended that the pilot's artificial horizon and the compass directional gyro in the Bristol 191 be fitted with potentiometers and pick-offs to provide autopilot signals. The instruments in the Whirlwind are not so fitted so a Sperry Horizon Gyro Unit and Direction Gyro from a G4B compass were installed in the main cabin.

As mentioned above, the signals of the H.G.U. are followed-up until the equipment is switched to the "Cruise" mode; the pick-off of the Directional Gyro is mechanically uncoupled from the gyro and centred with respect to its potentiometer until a heading signal is required.

3.7 Pilot's Controller (Figs 15 and 16)

In the first installation this took the form of a miniature stick (Fig.22) spring centred, which had the conventional movements for pitch and roll and was rotated about its axis to give yaw signals. It was mounted together with the trimmers, ahead and to the right of the pilot. For reasons given in Section 4, this form of controller was abandoned. The cyclic control in the left-hand cockpit of the Whirlwind was modified by fitting a movable grip, spring restrained and capable of about $\pm 2^{\circ}$ movement with respect to the rest of the stick; potentiometers and pick-offs were incorporated, full signal being obtained for a force of about 3-4 lb. A similar arrangement was fitted in the rudder circuit but with stronger springs and a dash-pot, found necessary for internal loop damping. In view of the small movement permitted the system is effectively a force-control one; the controls are however coupled to the autopilot servos at all times during engagement and partake in the stabilising movements made by them. These are not so disturbing to the pilot as might perhaps be expected, and it has proved possible to adjust the system to give satisfactory feel without undue autopilot feedback.

4 The Flight Development and Testing of the Type K Autopilot

4.1 Introduction

This section describes the flight development and testing which has been carried out to date in a Whirlwind H.A.S.22 (Sikorsky S55) helicopter W.V.202 (Fig.7) loaned to R.A.E. for the purpose by the Royal Navy. The pilot during these trials was Flt/Lt. J.I.T. Williamson, whose wide experience and enthusiastic co-operation have contributed greatly to the progress of the development.

So far about 120 hours flying have been carried out in the course of the work, mainly at Farnborough: this total includes about 25 hours whilst the aircraft was at R.N.A.S. Gosport during which time, as has already been mentioned, the experimental equipment was operated under Service conditions by the Royal Navy. This flying was the subject of a separate report and, resulting from the success of these trials, the Navy has decided to equip six Whirlwind aircraft of an operational squadron with an "engineered" version of the experimental equipment for extended Service trials.

The flying programme can be roughly sub-divided into three main phases:-

(a) The initial testing and stability investigations.

(b) System development followed by Navy Trials.

(c) Initial experiments on the automatic control of the aircraft in plan position whilst hovering over a sonar buoy.

The programme will be described in detail under the above headings.

4.2 The Initial Trials and Stability Investigations

Following installation and ground testing of the equipment at the Westland Aircraft Company's works at Yeovil the first experimental flight was carried out at Yeovil (with an R.A.E. pilot) on May 11th 1954. On this first flight reasonably stable performance was achieved even in hovering flight with very roughly estimated control parameter settings; following this encouraging start the aircraft was flown to Farnborough on the following day to continue the flight programme.

It is useful here to describe the basic operation of the autopilot in its earliest form. Whilst the electrical and mechanical conceptions of the servo system have changed little during the flight trials the functional aspect of the system has been the subject of continual development. In its original form the autopilot provided three different control modes for the pilot who was able to fly the aircraft through the autopilot using a "miniature stick" remote controller.

The three control modes were:-

(a) Stabiliser - In all axes aircraft control (i.e. cyclic stick or yaw pedal) was applied proportional to angular rate of the fuselage about the appropriate axis. The pilot's controller precessed the rate gyros so that a rate of rotation of the aircraft was produced proportional to controller stick displacement.

(b) Cruise - In all axes the rate gyro signal was integrated by means of an integrating motor system, and control was applied (within limits imposed by system threshold and drift) proportional to angular rate and angular displacement of the fuselage.

The pilot's controller function remained unaltered.

(c) Hover - Control law as in cruise but the controller signal was injected in the pitch and roll axis into the magnetic amplifier instead of into the gyro. This had the effect of producing an aircraft attitude (as defined by the fuselage) proportional to controller displacement. In yaw the controller still produced a rate of response of the aircraft.

It should be noted that although the names of the various control modes have remained unaltered their functions have changed considerably during the development.

The first flights as described above showed that the system appeared capable of stabilising the helicopter in hovering and in forward flight but certain shortcomings were apparent. It was noted, that on switching to the "Cruise Mode" i.e. when an angular position term was added to the control law the stability was reduced. In fact whilst the roll channel was still stable the pitch channel became unstable: even with the position signal reduced to a very low value it appeared impossible to achieve reasonable control in pitch. The first circuit change was therefore to introduce in the pitch and roll channels a condenser-resistor phase-advance network in the gyro signal line to obtain, at the controlled aircraft frequency observed, about 30° phase advance of gyro signal in pitch and rather less in roll.

In the following flights it was found that the stability was improved; more rate signal could be used with resulting tighter control but on the introduction of any position signal in pitch instability again occurred; however the oscillation was now only slowly divergent (previously it was a rapid divergence). There being no theoretical reason for the instability it was decided that a new system of integrating the rate gyro signal (to obtain the position signal) would be tried; by this time it was suspected that the integrating motor was not producing a satisfactory position signal and that this was the cause of the instability. A known circuit was tried in which a condenser (500-1000 μF) is inserted in the gyro spring coil line. With the condenser shunted by a large resistance (about 20 kilohms) a "leaky integrator" is produced and this was the circuit used (Fig.2).

When this change was tried in the pitch circuit an immediate improvement resulted and the roll circuit was accordingly modified in a similar way. It appeared by the end of June 1954, that the development was on the right lines and that further careful investigation of circuit constants would produce optimum stability.

At the same time it was felt that the stability was good enough to justify attempts to improve the functional performance of the equipment. It had for instance become apparent that the usefulness of the system as an aid in cruising flight was being reduced by the large and variable drifts (changes of datum) occurring. These were due partly to the bad centring of the pilot's controller and partly to rate gyro deficiencies. In an attempt to alleviate the trouble various drift compensating circuits were tried without success and it was eventually realised that the major cause of drift was in the controller. The situation was improved by careful adjustments and modification to this component.

Following these experiments it became possible to set up the equipment for optimum performance in all three control channels and this phase was completed by the end of July 1954.

4.3 System Development

During the early flight trials described above the equipment used was largely experimental in conception, and had been designed and installed in the aircraft in a way to facilitate the investigation in flight of the

stability problems. However in the course of the trials ideas began to take shape on a final form of the equipment suitable for operational use by Service Pilots.

Whilst some system development had taken place during the flying described above, the next phase of the flight trials which followed on naturally was devoted almost exclusively to this aspect. This phase lasted from early August to mid October and the way in which the system was built up during this time can now be described. The major features which evolved will be discussed in the order in which they were added to the system.

4.31 Yaw Compensation

In order to compensate for the effect of varying rotor torque a signal derived from the position of the collective pitch control was fed into the autopilot yaw channel. This relieved the pilot of the necessity of making large yaw control demands with changing flight conditions. In particular it became possible to make large changes in collective pitch and power settings whilst hovering (i.e. go up and down) without any significant change in heading. In manual flight this is particularly a problem in single rotor helicopters and in the Whirlwind the large yaw pedal forces involved are a major cause of pilot fatigue.

4.32 Attitude and Heading Monitors

As originally conceived, the equipment, in addition to providing artificial stability for the helicopter, should without monitoring instruments be capable of use as a "pilot monitored" autopilot for cruising flight. The intention was that pitch roll and yaw attitude signals would be obtained by integrating the rate gyro signals and that the drift level would be sufficiently constant and small to make it necessary for the pilot only occasionally to correct the flight path of the aircraft. Trimmers were provided to help to compensate for drift. However the drift levels proved to be rather high and variable, so it was decided to introduce into the control law for cruising flight attitude and heading signals, derived from pick-offs in the pilot's flight instruments, i.e. Artificial Horizon and Compass gyro. Units designed for the Mk.11 Automatic Pilot were modified to suit the helicopter system.

The heading signal was also found to be of value if used alone when hovering. It is common practice to maintain a fixed heading (generally into wind) when hovering; with the addition of the heading signal as well as the yaw compensation described above it became possible to maintain a sufficiently accurate heading unaffected by changes of power, height and plan position; a useful contribution to the hovering problem.

4.33 Pilot's Controls

The pilot's controller originally used, was adopted largely as an interim measure to start the test flying whilst ideas on the ultimate form of the controller were being formulated.

From the beginning of the project doubts had been expressed by helicopter pilots as to the advisability of using a "miniature stick" controller remote from the main flying controls. The objections were based mainly on safety considerations. Because the helicopter is an unstable aircraft and because the system was required for use in hovering flight near the ground, it was felt that in the event of an autopilot failure a dangerous condition could be reached before the pilot could regain control through his main flying controls.

The requirements for a safe system were thought to be (i) pilot's controllers to be incorporated in the cyclic stick and rudder pedals, and (ii) the cyclic stick etc. to remain coupled to the main rotor and tail rotor controls during autopilot operation.

In the initial flying so far described it was found that experienced pilots could quickly learn to fly the aeroplane using the miniature controller, performing all normal manoeuvres including take-off and landing. A certain amount of confidence was required however, and generally it was felt that without a safety pilot in the second seat the system was not acceptable for general use.

The most reasonable approach to the controller problem was thought to be the use of force sensitive pick-offs in the cyclic pitch and yaw controls so that pressure on the cyclic stick or yaw pedals produced demand signals for the autopilot. A suitably modified cyclic stick and yaw control link were installed in the aircraft in September and flight tested. Some difficulties were expected with this system, and they did in fact arise. The mounting of the force pick-off on a member (e.g. cyclic stick) which is moved in response to its signals results in a coupling effect which causes a "juddering" of the control, particularly when a constant demand signal is required. However steady control deflections are in general only needed in such manoeuvres as turns since trimmers are provided in the system to cope with constant trim demands in steady flight. In practice it was found that the level of the juddering could be reduced by a suitable design of the components (e.g. a hydraulic damper was added to the yaw force-sensitive link) and by adjustment of the circuit constants to a level acceptable to pilots once they had realised that a slightly different flying technique was needed. With the combination of this new control system and the stabilisation provided by the autopilot the helicopter became a stable machine which could be flown "hands-off" throughout its range of operating conditions and which could be manoeuvred apparently normally through the usual controls. The technique of flying became in fact to hold the controls extremely lightly and to apply a force only when a change in attitude was needed, this being analogous to a dynamically stable fixed wing aeroplane.

Whilst it is realised that this system of feeding in pilot's demand signals is not an ideal one, it appears that by suitable design and adjustment it can be made acceptable for general use and it has therefore been chosen. Any alternative systems are more complicated and have their own development problems.

It is of interest to note that the original "hover" mode controller function was abandoned at this stage. It had been thought that by providing a control giving aircraft angle (instead of rate) proportional to stick displacement it would be easier for a pilot to perform the task of positioning the machine over a point on the ground. In practice the control was found to be unpopular. This was partly due to the fact that it was unusual (a rate control approximates to a normal aircraft control law) and partly because of a conscious mental effort required to decide on how to use the controller when it could provide two different functions depending on the autopilot mode in use. Therefore in the adoption of "main stick" control the scheme was given up.

4.35 General Development

During this stage the final form of the control laws and controller functions were decided for the various modes of operation. The appropriate switching facilities were incorporated into redesigned switch units etc.

Some engineering development was carried out on various autopilot components; in particular some practical work was done to reduce the effect of the excessive aircraft vibration on the gyro units. It was found that a fairly stiff and well damped anti-vibration mounting was needed. Conventional "soft" mountings tended to amplify some of the low frequency vibrations found in the helicopter.

In concluding the description of this phase of the tests it is useful to describe the control facilities provided by the system in the various modes, the names of which have remained unchanged.

"Stabilise"

Cyclic Stick (Fore and aft and Lateral) applied proportional to fuselage rate + "short memory" position signal.

Yaw control proportional to fuselage rate.

Controller (Stick and Yaw pedals) give aircraft rates proportional to force.

"Cruise"

Attitude and Heading signals which are switched out or nulled during "stabilise" condition are added to the control law.

The Controller signals are switched out so that no inadvertent demands are put in and the autopilot maintains the attitude existing on switching to "cruise".

"Hover"

The conditions in the pitch and roll channels are as in "Stabilise", and the aircraft can be positioned using the cyclic stick.

The Yaw channel is as in "Cruise" so that a heading is maintained.

N.B. The Yaw compensator is used all the time. A "Manoeuvre Button" on the stick permits a temporary reversion to "Stabilise" from "Cruise" or "Hover" so that attitude or heading changes can be made without the use of the main switch unit.

A new set of equipment containing all the features described was installed in the aircraft early in October 1954 and the experimental installation was tidied up. Following some setting up the equipment was tested by the Navy at R.N.A.S. Gosport during the last two weeks in October (section 4.1).

4.4 The Initial Experiments in Automatic Hovering

As has been mentioned previously the complete helicopter autopilot requirement includes the provision of fuller automatic pilot facilities such as co-ordination of turns, automatic means of hovering over a dipping sonar and of coupling to navigation systems.

The system so far described, consisting of a stabiliser plus a "manoeuvre holding" autopilot, is however in itself a worthwhile aid to a helicopter pilot. It enables instrument flying to be carried out with comparative ease and, by means of the added stability and heading holding feature, simplifies the pilot's task in sonar-dipping. Therefore the proposal is that this system shall form the basic autopilot for early production; it will be possible to add more facilities as they become available.

The first of these to be investigated was the automatic control of the helicopter in the plan position whilst hovering over the sonar-buoy. As has been described measurements of cable-angle from the sonar winch gear are used as input signals in the autopilot pitch and roll channels.

The first flight experiments with this system took place from November 1st, 1954, to April 16th, 1955, when flying ceased temporarily for a routine aircraft inspection. The amount of flying involved was rather limited as during this time the equipment had been modified to take the additional components needed. This had involved a slight re-arrangement of the basic autopilot circuits to provide the new control mode "Auto-hover", and the construction of the necessary sonar coupler unit. In addition, a novel form of integrating accelerometer has been developed whose signals provide the damping terms in the position control loop; this is described more fully in Appendix II.

The stability problems have been largely solved using the system described above, i.e. cable-angle signals plus the speed term obtained from the integrating accelerometer (gravity compensated). It was found necessary to modify the existing potentiometers in the sonar gear which provide cable angle indications to the pilot, as the original system provided too small a voltage swing for the autopilot. The new potentiometers however still operate the visual indicators in addition to the autopilot.

In passing it should be mentioned that attempts were made to use differentiated cable angle signals for damping but no success was achieved.

The state of the experiments when flying was halted was that the stability was fairly satisfactory and only a little more accurate setting up of the autopilot parameters appeared necessary. Trouble was being experienced with a constant drift (resulting in a dragging of the buoy) but this was due to instrumental faults which have now been found and rectified.

It is proposed to try an integral of cable angle signal in the pitch channel to eliminate steady state errors due to surface wind. The equipment is normally set up for the hovering case with a 15 kt wind and the appropriate aircraft C.G. position for the Sonar role.

The auto-hovering experiments will be described more fully in a separate note.

4.5 Further Work

When flight experiments start again the auto-hovering work will be continued.

During the aircraft overhaul two additional autopilot servomotors have been installed, one for collective pitch and one for engine throttle control. The necessary control boxes have been designed and when completed will be installed in the aircraft so that experiments in control of height while hovering and automatic control of rotor speed can be carried out.

A barometric height control for cruising flight will also be tried.

Finally it is proposed that experiments in the automatic co-ordination of manoeuvres and the provision of more advanced autopilot functions will be tried.

5 Conclusions

Although the present note is an interim statement on the work being done on auto-control of helicopters, certain conclusions may usefully be drawn at this stage, viz.

(1) A simple artificial stabilisation system can be provided for helicopters using rate-gyros as basic signal elements. The installed weight of the present experimental equipment (stabiliser plus manoeuvre holding autopilot) is 60-65 lb excluding cable harness. For other machines it will depend on the servo-motors required; where power controls are already fitted some reduction may be possible. Improved circuit techniques, e.g. use of transistors, may also effect some weight saving as will some mechanical redesign now being carried out for the Service Trials equipment.

(2) Auto-control operation through the main stick, whilst retaining direct coupling to the rotor controls, is feasible and acceptable.

(3) The provision of a Stage 1 (section 1) equipment with the addition of the yaw compensator and a heading monitor is a worthwhile contribution to the task of Sonar operation. Its usefulness in cruising flight is limited at present by the performance of available rate gyros. The provision of suitable monitors enables adequate cruising performance to be realised at the expense of slight extra weight, using signals obtained from modified pilots flight instruments (provided these are electrical types).

REFERENCE

No.	Author	Title, etc.
1	Staffs of Auto Controls Section I.A.P. Dept. and Tele-Control Section, Radio Dept.	Autopilots Type B and C Tech. Note IAP 1001

Attached: List of Symbols to Appendices
Appendices I and II
Drg Nos IN 32585 to 32590, 30409
Neg Nos 121,739-121,746
Detachable Abstract Cards

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List of Symbols in Appendices

η	fore-and-aft tilt of normal to rotor plane (assumed thrust-line) relative to line joining rotor hub to C.G. of helicopter: positive backwards
ξ	lateral tilt of normal to rotor plane etc: positive to right, looking forward
u	increment of forward speed
v	increment of sideways speed; positive to right, looking forward
x	fore-and-aft displacement from datum in hovering: $x = \int u\,dt$
θ, ϕ	attitude and bank angle of fuselage as in normal aircraft convention
$W = mg$	weight of helicopter
k_x, k_y	radii of gyration about roll and pitch axes respectively
h	distance from rotor hub to C.G. of helicopter
$M_\eta = \frac{1}{mk_y^2} \cdot \frac{\partial M}{\partial \eta}$	derivative of angular acceleration in pitch with respect to η
$L_\xi = \frac{1}{mk_x^2} \cdot \frac{\partial L}{\partial \xi}$	derivative of angular acceleration in roll with respect to ξ
a_u	derivative of rotor tilt with respect to u
a_v	derivative of rotor tilt with respect to v
a_q	derivative of rotor tilt with respect to pitching angular velocity of fuselage
a_p	derivative of rotor tilt with respect to rolling angular velocity of fuselage
η_s, ξ_s	applied cyclic control, fore-and-aft and lateral respectively
a_{qa}	coefficient of $\dot{\theta}$ in autopilot control equation
a_θ	coefficient of θ in autopilot control equation
a_x	coefficient of x in autopilot control equation
λ	general root of characteristic equation for stability
p	Heaviside-Laplace operator

APPENDIX I

Elementary Stability Theory of the Helicopter

1 Hovering

(a) Uncontrolled Stability

The peculiarities of control in the helicopter arise from the fact that both moments and forces are directly produced by tilting one control, viz. the rotor. Moments are produced by changes in rotor tilt relative to the fuselage, horizontal forces by changes in tilt relative to the vertical. To keep in line with conventional aircraft notation, the lateral and fore-and-aft tilts of the rotor thrust line relative to the line from rotor hub to aircraft C.G. will be known as ξ and η respectively. Ignoring thrust changes, the angular acceleration in pitch caused by a tilt η is

$$\frac{Wh\eta}{mk_y^2}$$

where $W = mg$ is the weight of the machine, h the distance between rotor hub and C.G., k_y is the radius of gyration about the y axis.

The parameter $\frac{gh}{k_y^2}$ will be called M_η, and $\frac{gh}{k_x^2}$ correspondingly L_ξ.

Considering the case of pitch (roll is essentially similar), the following equations hold (Fig. 24):

$$\dot{u} = -g(\theta + \eta) \tag{1}$$

$$\ddot{\theta} = \eta M_\eta . \tag{2}$$

The speed u is assumed to be small.

A third equation is provided by the relation between η_s, the cyclic control input, η, u and θ. There are two major factors in this relation:

(i) Backward tilt of the rotor-plane with forward speed

As a result of the forward motion of the rotor hub through the air the lift on the advancing blades is greater than that on the retreating blades, due to the speed difference. This gives rise to a backward tilt of the tip-path plane until the resulting redistribution of incidence (decrease on the advancing blades and increase on the retreating blades) cancels the difference in lift. The angle of tilt can be shown to be proportional to the helicopter forward speed: i.e.

$$\eta_u = a_u . u .$$

Furthermore, since the rotor hub is distant h from the C.G. of the aircraft, angular motions about the C.G. give translational movements of the hub, the speed being $-h\dot{\theta}$; hence a contribution

$$\eta_u' = -a_u h\dot{\theta} .$$

(ii) Lag between angular motions of the fuselage and the rotor-plane

The inertia of the rotor gives rise to the above lag the amount being $-a_q \cdot \dot{\theta}$.

The general equation for η, the fore-and-aft tilt of the thrust line relative to fuselage datum is then:

$$\eta = \eta_s + a_u u - (a_q + a_u h) \cdot \dot{\theta} \quad . \tag{3}$$

Similarly

$$\xi = \xi_s - a_v \cdot v - (a_p + a_v h)\dot{\phi} \quad . \tag{3a}$$

Using equations (1) and (2) to eliminate η we get

$$\frac{\dddot{\theta}}{M_\eta} = \eta_s - ga_u \int \left(\theta + \frac{\ddot{\theta}}{M_\eta}\right) dt - a'_q \dot{\theta} \; ; \; a'_q = a_u h + a_q$$

$\therefore$

$$\ddot{\theta} + (ga_u + a'_q M_\eta) \dot{\theta} + ga_u M_\eta \int \theta \cdot dt = \eta_s M_\eta \quad . \tag{4}$$

This leads to the characteristic equation of the uncontrolled motion

$$\lambda^3 + (a_u g + a'_q M_\eta) \lambda^2 + a_u g M_\eta = 0 \tag{5}$$

which although the coefficients are positive, corresponds to an essentially unstable motion, as may be seen by considering that the factors of such a cubic, written as:

$$(\lambda + \alpha) (\lambda^2 + \beta\lambda + \gamma)$$

require

$$\alpha\beta + \gamma = 0$$

for the coefficient of λ to vanish; since, in fact, α and γ are both positive, β must be negative, implying a divergent oscillation. For stability the coefficient of λ in the cubic must be positive and not zero; in fact, by putting $\beta = 0$, it is clear that the coefficient of λ in the cubic must exceed γ for positive damping.

(b) Controlled Stability

The effect of the autopilot is introduced by the relation between η_s, the control movement applied, and the controlled quantities θ, $\dot{\theta}$ etc. Since stability requires the presence of a term in λ in the cubic equation, which corresponds to a term in θ in the differential equation, a useful control would be:

$$\eta_s = -a_\theta \theta - a_{qa} \dot{\theta}$$

which leads to an equation:

$$\ddot{\theta} + \left[ga_u + (a_q' + a_{qa})M_\eta\right]\dot{\theta} + a_\theta M_\eta \cdot \theta + a_u g M_\eta \int \theta \cdot dt = 0 . \qquad (6)$$

The $\dot{\theta}$ term in the control law is necessary since the θ term stabilises the divergent oscillation at the expense of the subsidence, and the inherent damping of the helicopter is insufficient to provide adequately for both.

Approximate values of a_u, h and a_q for the Whirlwind are

$$a_u = 0.3 \times 10^{-3}, \qquad a_q = 0.037, \qquad h = 8 \text{ ft}$$

$$a_q' = a_q + a_u h = 0.039 \approx 0.04$$

$$ga_u = 0.0096 \approx 0.01 .$$

To obtain M_η, the radius of gyration of the helicopter must be estimated, and little information is available on the point. Using the dimensions of the machine, the radius of gyration in pitch would appear to be of the order of 7 to 9 ft, these being the values for a solid and hollow box respectively of the same total mass, with a tail added, containing 0.1 of the total mass. Thus, the convenient value of 8 ft is not unreasonable, giving $M_\eta = \frac{gh}{k_y^2} = 4$. The value of k_x is estimated at 5 ft, giving $L_\xi = 10$.

The equation with these values becomes (for pitch)

$$\lambda^3 + 0.17\lambda^2 + 0.04 = 0 = (\lambda + 0.41)(\lambda^2 - 0.238\lambda + 0.0975)$$

for the uncontrolled aircraft.

With the control terms, we have

$$\lambda^3 + (0.17 + a_{qa} M_\eta)\lambda^2 + a_\theta M_\eta \lambda + 0.04 = 0$$

i.e.

$$\lambda^3 + (4a_{qa} + 0.17)\lambda^2 + 4a_\theta \lambda + 0.04 = 0 .$$

For large autopilot gearings i.e. large values of a_{qa} and a_θ, this equation splits into the approximate factors:

$$\left(\lambda + \frac{0.04}{4a_\theta}\right)(\lambda^2 + 4a_{qa}\lambda + 4a_\theta) = 0 .$$

There are two points to be noticed about the factorised equation. Firstly, there is a poorly damped subsidence $\left(\lambda + \frac{0.04}{a_\theta}\right) = \left(\lambda + \frac{a_u g M_\eta}{a_\theta}\right)$. The larger a_θ, the smaller the damping implied by this root, the values of

a_u and M_η being characteristics of the aircraft. If, however, one could introduce a speed term in the control equation, the effective value of a_u would be changed and might be made more advantageous. This poorly damped mode is not of great importance when pilot monitored flying is undertaken or manual sonar dipping; in these cases, the short period stability is the essential gain. For cruising flight however, a poorly damped subsidence or long-period oscillation may be a nuisance; furthermore, it will be shown that for automatic hovering, a "u" control is essential for stability. It is therefore proposed to develop such a control and try it for cruising flight as well as automatic hovering.

The second factor in the equation, the short period oscillation, has an undamped natural frequency $\frac{\sqrt{a_\theta}}{\pi}$, and damping ratio $\sqrt{a_\theta} \cdot \frac{a_{qa}}{a_\theta}$. For practical reasons, the ratio of $\frac{a_{qa}}{a_\theta} = 1 : 2$ is convenient to use and has been chosen for this autopilot. Hence the damping ratio $\approx \frac{1}{2} \cdot \sqrt{a_\theta}$, for values of a_θ exceeding about 0.5. Here again, some improvement is desirable, and the use of phase-advance, discussed below, makes it possible. Before showing this however, it is instructive to assess the effect of the present mode of obtaining an angular position signal with the electric-spring gyro and shunted condenser.

The use of the leaky integrator in the rate-gyro circuit modifies the control law somewhat from the form assumed above, and, using the Heaviside-Laplace notation, it becomes with the type of circuit used

$$\tau_s = - a_{qa}\left[1 + \frac{2n}{1+np}\right] p \cdot \theta = - a_{qa} \cdot \dot{\theta} - \frac{2na_{qa}}{1+np} \cdot \dot{\theta}$$

where n is the time constant of the integrator leak in seconds. The resulting stability equation is

$$\lambda^4 + \lambda^3\left(4a_{qa} + 0.17 + \frac{1}{n}\right) + \lambda^2\left(8a_{qa} + \frac{4a_{qa} + 0.17}{n}\right) + 0.04\,\lambda + \frac{0.04}{n} = 0 \ .$$

This is seen to reduce to the equation previously given as $n \to \infty$, remembering $a_\theta = 2a_{qa}$.

Taking $a_{qa} = 0.2$ as a typical value, the solutions for differing values of n are found to be

<u>$n = 5$</u> $$\left[(\lambda + 0.575)^2 + (1.2)^2\right]\left[(\lambda + 0.01)^2 + (0.066)^2\right]$$

<u>$n = 10$</u> $$\left[(\lambda + 0.524)^2 + (1.18)^2\right]\left[(\lambda + 0.011)^2 + (0.048)^2\right]$$

<u>$n = \infty$</u> $$\left[(\lambda + 0.472)^2 + (1.16)^2\right]\left[\lambda + 0.025\right] .$$

There is therefore some improvement in short period damping, but the main effect is to turn the slow subsidence into a long period oscillation, not very well damped, and whose damping becomes progressively worse as the time constant of the leak decreases, the frequency meanwhile increasing.

Using the values of n eventually chosen after flight trials, viz. $7\frac{1}{2}$ for pitch and $13\frac{1}{2}$ in. roll, the control equations become

$$\eta_s = -0.3\left[1 + \frac{15}{1 + 7.5p}\right] p \cdot \theta$$

$$\xi_s = -0.1\left[1 + \frac{27}{1 + 13.5p}\right] p \cdot \phi$$

the values $a_{qa} = 0.3$, $a_{pa} = 0.1$ also being settings found during flight trials.

The stability equations are:-

In pitch, $(M_\eta = 4)$

$$\lambda^4 + 1.5\,\lambda^3 + 2.58\,\lambda^2 + 0.04\,\lambda + 0.0053 = 0$$

$$= \left[(\lambda+0.75)^2 + (1.4)^2\right]\left[(\lambda+0.0073)^2 + (0.046)^2\right] .$$

In roll, $(L_\xi = 10)$

$$\lambda^4 + 1.47\,\lambda^3 + 2.1\,\lambda^2 + 0.1\,\lambda + 0.0074 = 0$$

$$= \left[(\lambda+0.71)^2 + (1.22)^2\right]\left[(\lambda+0.023)^2 + (0.083)^2\right] .$$

The damping is not particularly satisfactory for any of these modes, but the short period damping may be substantially improved by a phase-advance network in the gyro signal line. This comprises a resistance and condenser across the attenuator and, in the case of the roll channel, for example, gives a control equation approximately

$$\xi_s = -0.1\left[1 + \frac{27}{1 + 13.5p}\right](1 + 0.55p) \cdot p\phi .$$

The stability equation derived from this is:

$$\lambda^4 + 1.67\,\lambda^3 + 1.4\,\lambda^2 + 0.077\,\lambda + 0.005 = 0$$

$$= \left[(\lambda+0.81)^2 + (0.81)^2\right]\left[(\lambda+0.027)^2 + (0.055)^2\right] .$$

The short period frequency has been decreased and the damping improved for two reasons; the phase-advance of the rate component of the gyro signal is equivalent to an added inertia term in the equations; the phase-advance of the quasi-position signal gives added damping.

The equation for the control in pitch, including the phase-advance term, is approximately:-

$$\eta_s = -0.3\left[1 + \frac{15}{1 + 7.5p}\right](1 + 0.3p)\, p \cdot \theta$$

which leads to a stability equation

$$\lambda^4 + 1.67\,\lambda^3 + 1.90\,\lambda^2 + 0.029\,\lambda + 0.0039 = 0$$

$$= \left[(\lambda + 0.83)^2 + (1.09)^2\right]\left[(\lambda + 0.007)^2 + (0.046)^2\right] .$$

It will be observed that although the period of the slow oscillation is of the same order as that in roll, the damping is far worse.

2 Cruising flight

In cruising flight the helicopter is statically unstable in pitch, i.e. a change in incidence leads to a displacement of the rotor-plane, and hence a control moment, in the same direction. The effect is however, small and, as it is not known for the Whirlwind, no calculations have been made. In principle, an increase in the coefficient of θ in the control equation would compensate for the effect. In fact, the addition of monitor terms to the control in cruising flight works in this direction. The use of a speed control to improve the poor long-period damping in pitch is well worth considering in cruising flight, and work on this will be done.

APPENDIX II

Automatic Hovering

To achieve automatic hovering, one must introduce a signal giving the plan-position of the Helicopter with respect to ground, or sea, and this is conveniently obtained from the cable-angle indicator of the Sonar, which normally supplies this information to the pilot. Ignoring the practicalities of the problem, we wish to find what sort of signals will be necessary to attain stable control, and of what order of magnitude they must be. The simplest approach is to find the effect of a simple plan-position signal, i.e. one which is made to give cyclic control proportional to linear displacement of the machine. Considering the case of pitch, we have an additional control term proportional to $x = \int u\,dt$, or operationally, $x = \frac{u}{p}$.

From the equation (3) in Appendix I, i.e.

$$\eta = \eta_s + a_u u - a_q' \cdot \dot{\theta}$$

we can see that control terms in η_s proportional to u or $\dot{\theta}$ merely change the effective value of a_u or a_q', so that the stability equation for such a control is obtained by replacing in the uncontrolled stability equation (5) of Appendix I, e.g. a_u by $(a_u + a_{ua})$ and a_q' by $(a_q' + a_{qa})$ for a control in which

$$\eta_s = a_{ua} u - a_{qa} \dot{\theta} \quad .$$

To incorporate terms in x or θ, we use the relations $x = \frac{u}{p}$, $\theta = \frac{\dot{\theta}}{p}$ in the same way, so that for a control

$$\eta_s = a_x \cdot x = a_x \cdot \frac{u}{p} \; ,$$

the coefficient a_u is replaced by $a_u + \frac{a_x}{p}$, or, using λ instead of p, $a_u + \frac{a_x}{\lambda}$. The effect on stability of an "x" control, with no other signals, may be seen in this way to give an equation

$$\lambda^3 + \lambda^2 \left[g\left(a_u + \frac{a_x}{\lambda}\right) + a_q' M_\eta \right] + gM_\eta \left[a_u + \frac{a_x}{\lambda} \right] = 0$$

$\therefore$

$$\lambda^4 + (ga_u + M_\eta a_q')\lambda^3 + ga_x \lambda^2 + gM_\eta a_u \lambda + gM_\eta a_x = 0 \; .$$

The stability discriminant for this equation,

$$(g M_\eta a_u)^2 - g^2 M_\eta a_x a_u (ga_u + M_\eta a_q') + gM_\eta a_x (ga_u + M_\eta a_q')^2$$

$$= (gM_\eta a_u)^2 + gM_\eta a_x (ga_u + M_\eta a_q') M_\eta a_q' \; ,$$

is essentially positive, and the motion is therefore unstable.

The addition of control terms in $\dot{\theta}$ and u would merely change the effective values of a'_q and a_u, and, in view of the nature of the discriminant, could not confer stability. As before, a θ control must be added, and this gives an equation

$$\lambda^3 + \lambda^2\left[g\left(a_u + \frac{a_x}{\lambda}\right) + M_\eta\left(a'_q + \frac{a_\theta}{\lambda}\right)\right] + gM_\eta\left(a_u + \frac{a_x}{\lambda}\right) = 0$$

i.e.

$$\lambda^4 + \lambda^3(ga_u + M_\eta\, a'_q) + \lambda^2(ga_x + M_\eta\, a_\theta) + \lambda \,.\, gM_\eta\, a_u + gM_\eta\, a_x = 0 \;.$$

In this equation, all the "derivatives", a_u, a_q, etc. may be regarded as being obtained by appropriate autopilot signals, the aircraft's own derivatives being practically negligible.

The new discriminant is found to be:

$$M_\eta^2\left[(ga_u)^2 + g(ga_u + M_\eta\, a'_q)\;(a_x\, a'_q - a_u\, a_\theta)\right] \;.$$

As this must be negative for stability, it is clear that both a_u and a_θ must be sufficiently large compared with the other terms, but that whereas the effect of a_θ is always in the same sense, that of a_u may be detrimental if it is too large. We therefore conclude that for satisfactory automatic hovering, control terms in all the variables θ, $\dot{\theta}$, x and u are necessary.

It is not proposed to discuss further in this Note detailed stability calculations, which will be dealt with in a separate memorandum. An indication of the method of deriving the speed term may, however, be of interest. There are two obvious possibilities at the outset, viz. (i) differentiation of the cable angle signal from the Sonar and (ii) a dynamical method. Method (i) would have the advantages of relative simplicity and of maintaining a constant ratio between the rate and position terms irrespective of cable length, but it is open to the objection that it would probably give very "noisy" signals in anything but almost calm conditions. It was not seriously considered therefore, and method (ii) was chosen. This involves the use of gravity-compensated integrating accelerometers in the fore-and-aft and lateral planes. A convenient way of achieving the required result is to use the electric-spring principle as in the rate-gyros, the gyro rotor being replaced by an unbalanced mass (Fig.23). The arm carrying the mass rotates about a vertical axis; the arm of the fore-and-aft accelerometer lies athwartships, that of the lateral accelerometer, fore-and-aft. They both carry pick-offs and potentiometers, together with torque coils and magnets essentially the same as those of the rate gyros. Considering the fore-and-aft instrument, there will be a torque τ on the arm, where

$$\tau = m\ell\,(\dot{u} + g\theta)$$

m = unbalanced mass

ℓ = effective length of arm

(for small angles) .

Thus, with the electric-spring circuit in operation, the pick-off will set itself to give a current

$$i_F \quad \propto \quad \mathring{u} + g\theta \ .$$

By using a second coil, however, fed from the vertical gyro, and therefore carrying a current proportional to θ, the part of the torque required to balance the effect of tilt, viz. $m\ell g\theta$, can be supplied by this coil, leaving, for the electric-spring coil,

$$i_F \propto \mathring{u} \ .$$

By including in the electric-spring circuit a condenser, this current is integrated and we obtain a pick-off voltage proportional to u and $\mathring{u}$; the condenser may be shunted to make the signal a transient one, as it is required for stabilisation only. It will be noticed that this technique gives an acceleration ($\mathring{u}$) signal as well as a speed term, with corresponding modifications to the stability.

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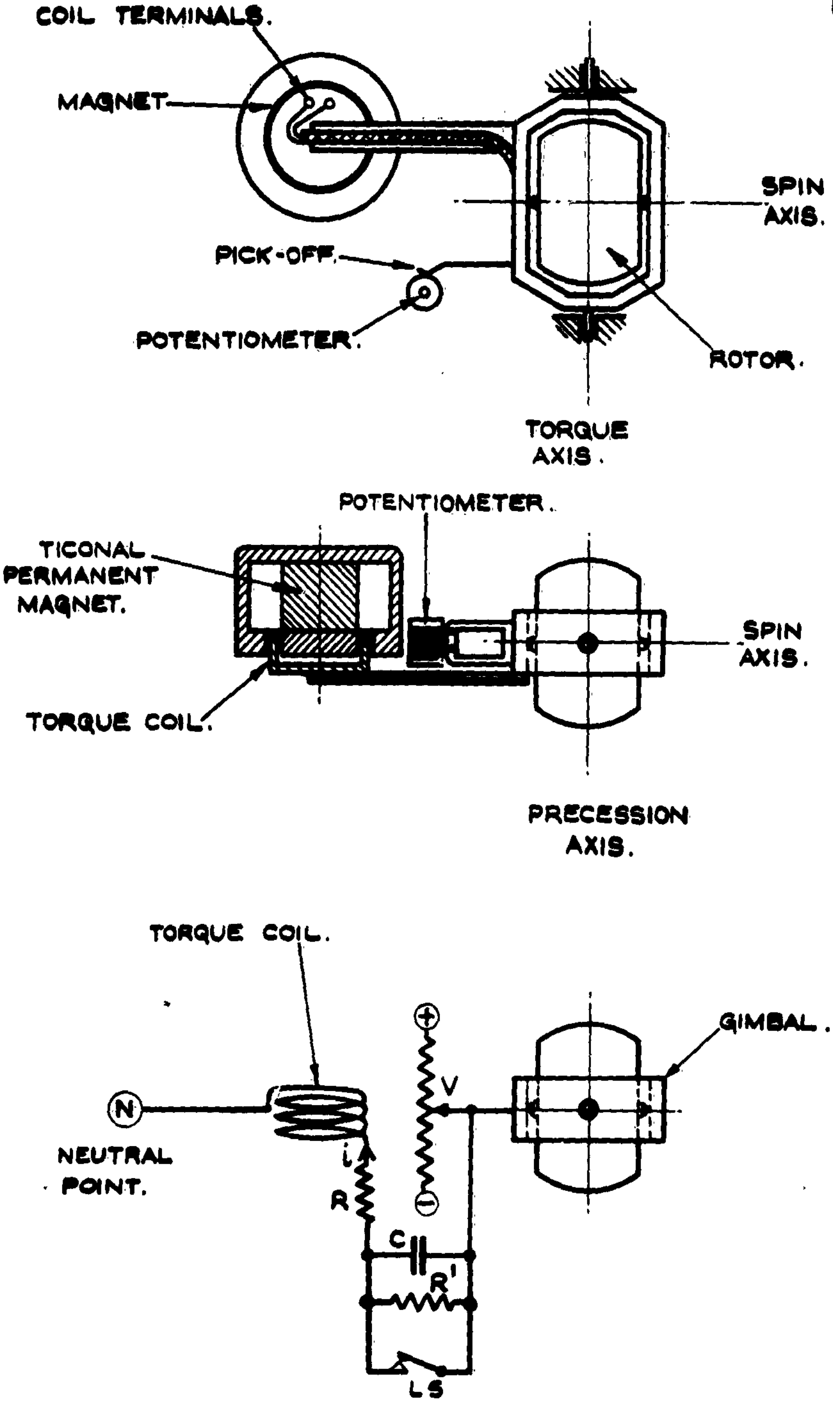
FIG. I.



$$i = K\dot{\theta}$$

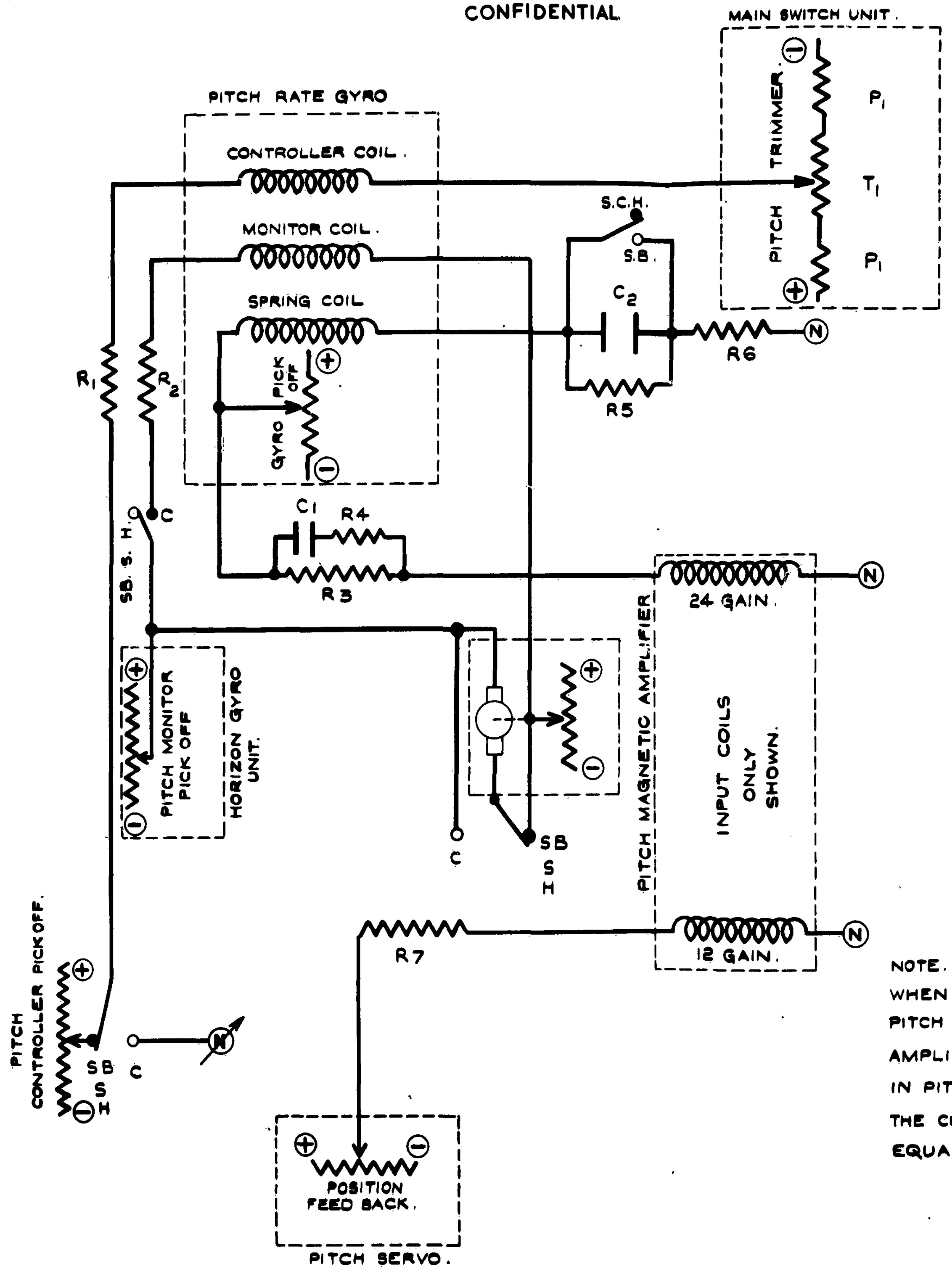
$$V = iR = Rk\dot{\theta} \quad \text{(SWITCH CLOSED)}$$

$$V = k\left[R + \frac{R'}{1 + pcR'}\right]\dot{\theta} \quad \text{(SWITCH OPEN)}$$

FIG. I. RATE GYRO. (SCHEMATIC.)

NOTE: MONITOR & CONTROLLER COILS NOT SHOWN.

FIG. 2.



MAIN COMPONENT VALUES.

	PITCH	ROLL
R_1	220 Ω	100 Ω
R_2	4·7K Ω	3·9K Ω
R_3	4·7K Ω	4·7K Ω
R_4	470 Ω	470 Ω
R_5	15K Ω	27K Ω
R_6	0 Ω	0 Ω
R_7	1·8K Ω	560 Ω
P_1	150 Ω	150 Ω
T	50 Ω	50 Ω
C_1	64 μF	100 μF
C_2	500 μF	500 μF

SYMBOLS

S.B.	STAND BY
S	STABILISE
C	CRUISE.
H	HOVER
⊕ ⊖	28 VOLT D.C. SUPPLY
(N)	FIXED NEUTRAL POINT.
(N) with arrow	ADJUSTABLE NEUTRAL POINT.

NOTE.
WHEN USED ORIGINALLY THE MONITOR SIGNALS IN PITCH AND ROLL WERE FED INTO A 6 GAIN MAGNETIC AMPLIFIER COIL, WHEN R_2 HAD THE VALUE 1·8K Ω IN PITCH AND 1·5K Ω IN ROLL. THIS DIAGRAM SHOWS THE CURRENT PRACTICE. BOTH SCHEMES SEEM EQUALLY EFFECTIVE.

FIG. 2. BASIC SIGNAL CIRCUIT DIAGRAM. (PITCH & ROLL.)
(N.B. PITCH ONLY SHOWN.)

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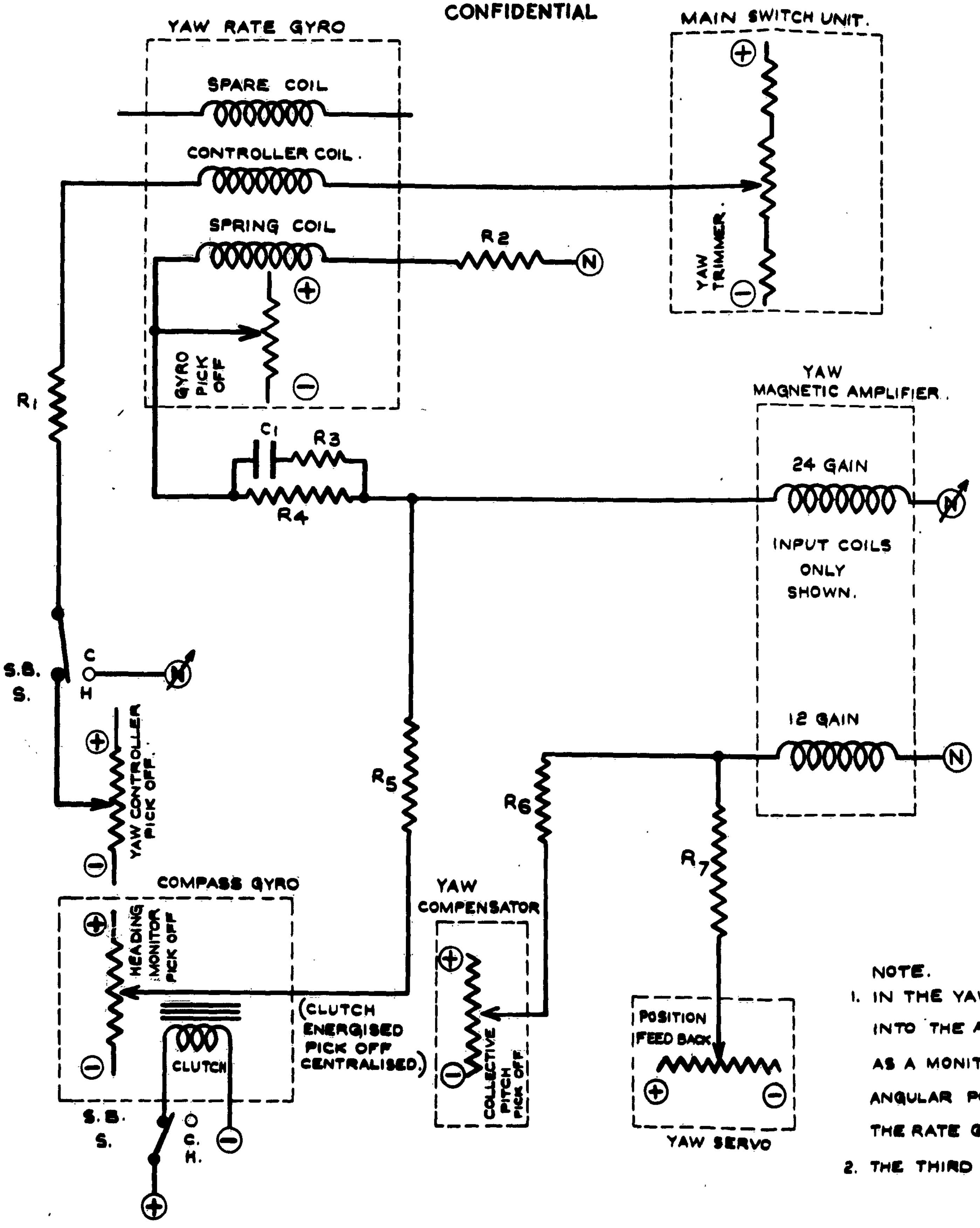
FIG. 3.



MAIN COMPONENT VALUES

R_1	100 Ω
R_2	0 Ω
R_3	100 Ω
R_4	1K Ω
R_5	1·5K Ω
R_6	1K Ω
R_7	1K Ω
T	100 Ω
P_1	100 Ω
C_1	100 μF

SYMBOLS

S. B.	STAND BY.
S.	STABILISE.
C.	CRUISE.
H.	HOVER.
⊕ ⊖	28 VOLT D.C. SUPPLY.
(N)	FIXED NEUTRAL POINT.
(N)	ADJUSTABLE NEUTRAL POINT.

NOTE.

1. IN THE YAW CHANNEL THE HEADING SIGNAL IS FED INTO THE AMPLIFIER AS A MAIN CONTROL TERM NOT AS A MONITOR. THERE IS NOT, AS IN PITCH AND ROLL, AN ANGULAR POSITION SIGNAL OBTAINED BY INTEGRATING THE RATE GYRO SIGNAL.
2. THE THIRD GYRO COIL IS NOT USED AT PRESENT.

FIG. 3. BASIC SIGNAL CIRCUIT DIAGRAM. (YAW.)

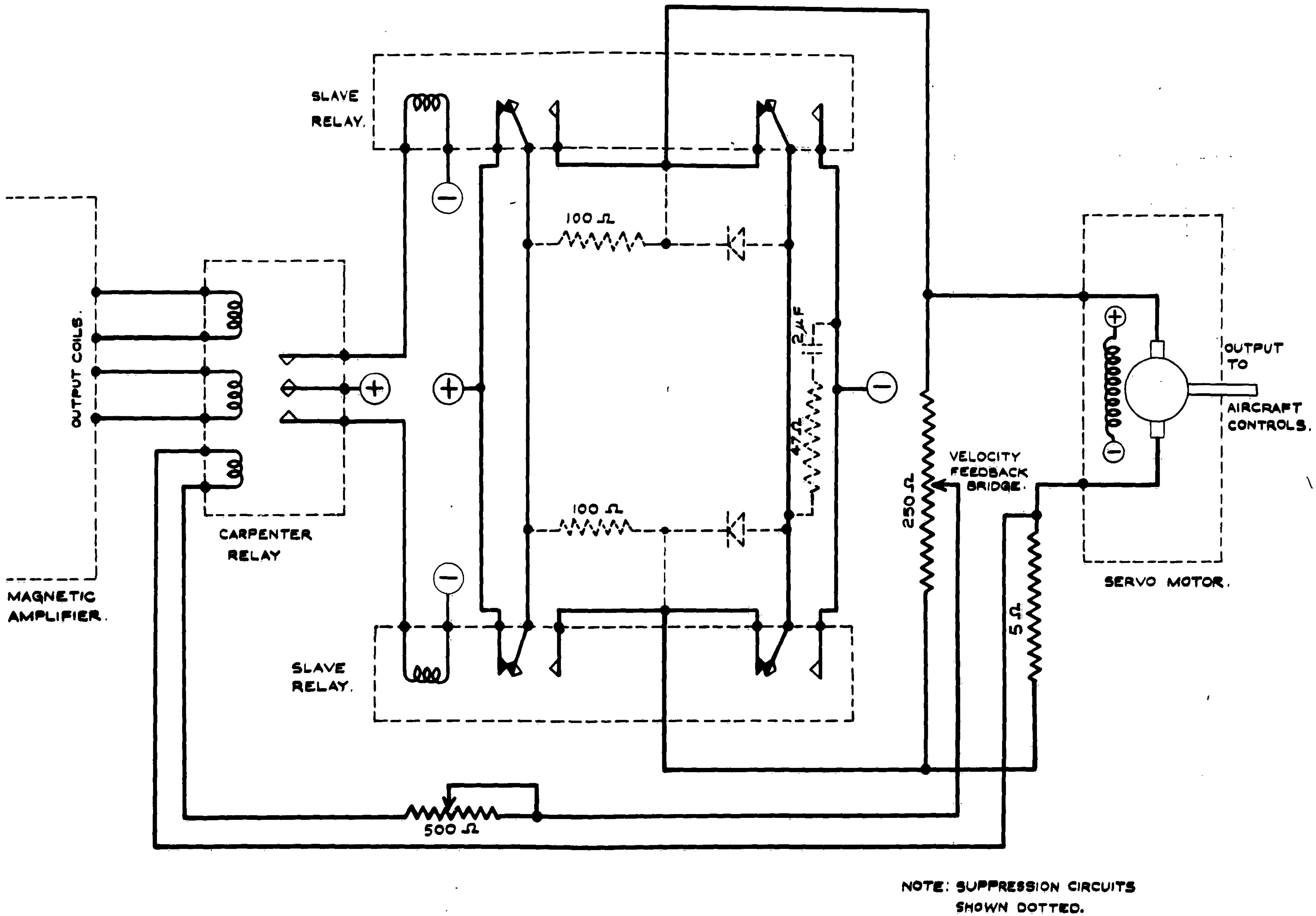


FIG. 4. BASIC SERVO CIRCUIT DIAGRAM. PITCH & ROLL.

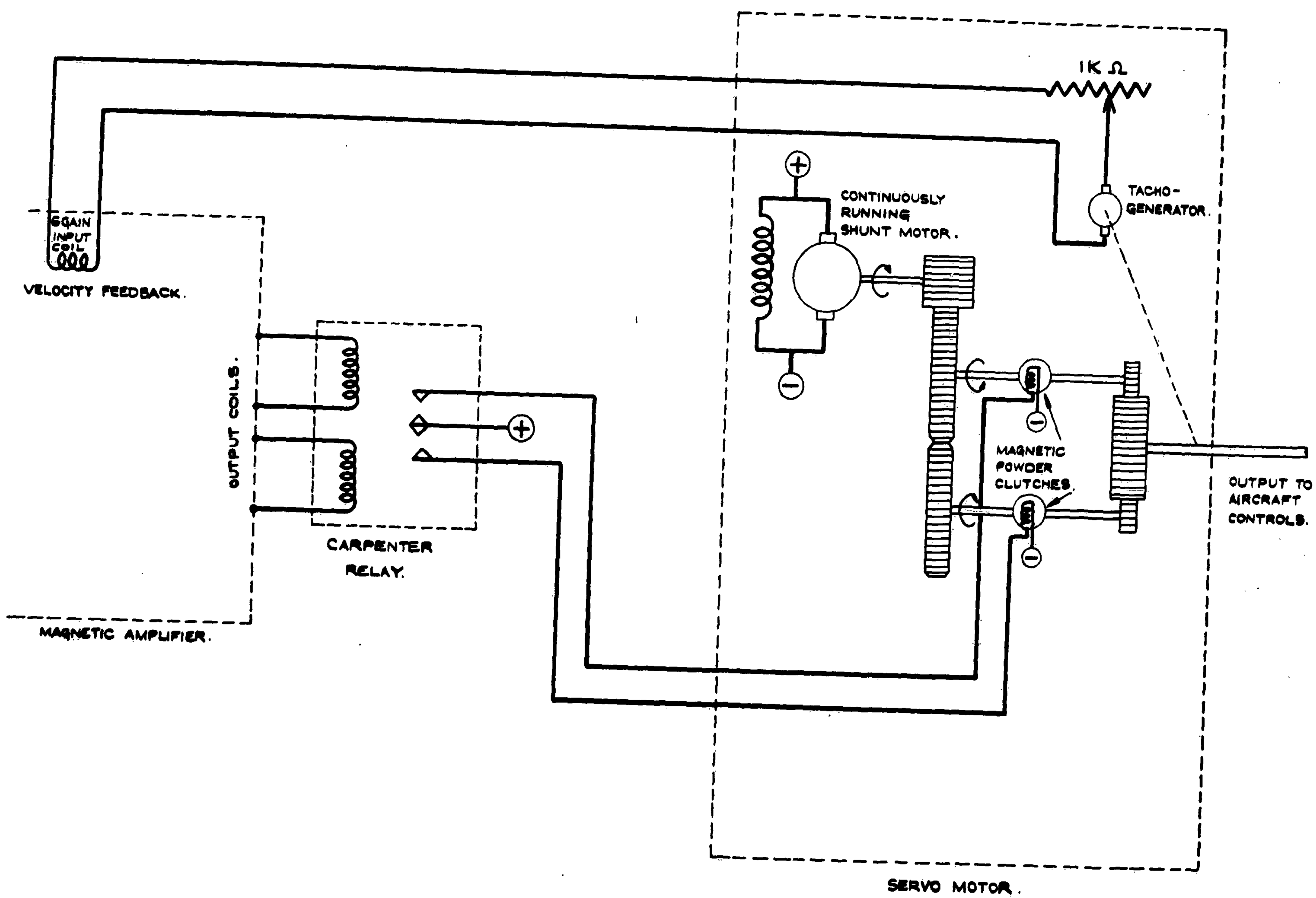


FIG. 5. BASIC SERVO CIRCUIT DIAGRAM. YAW.

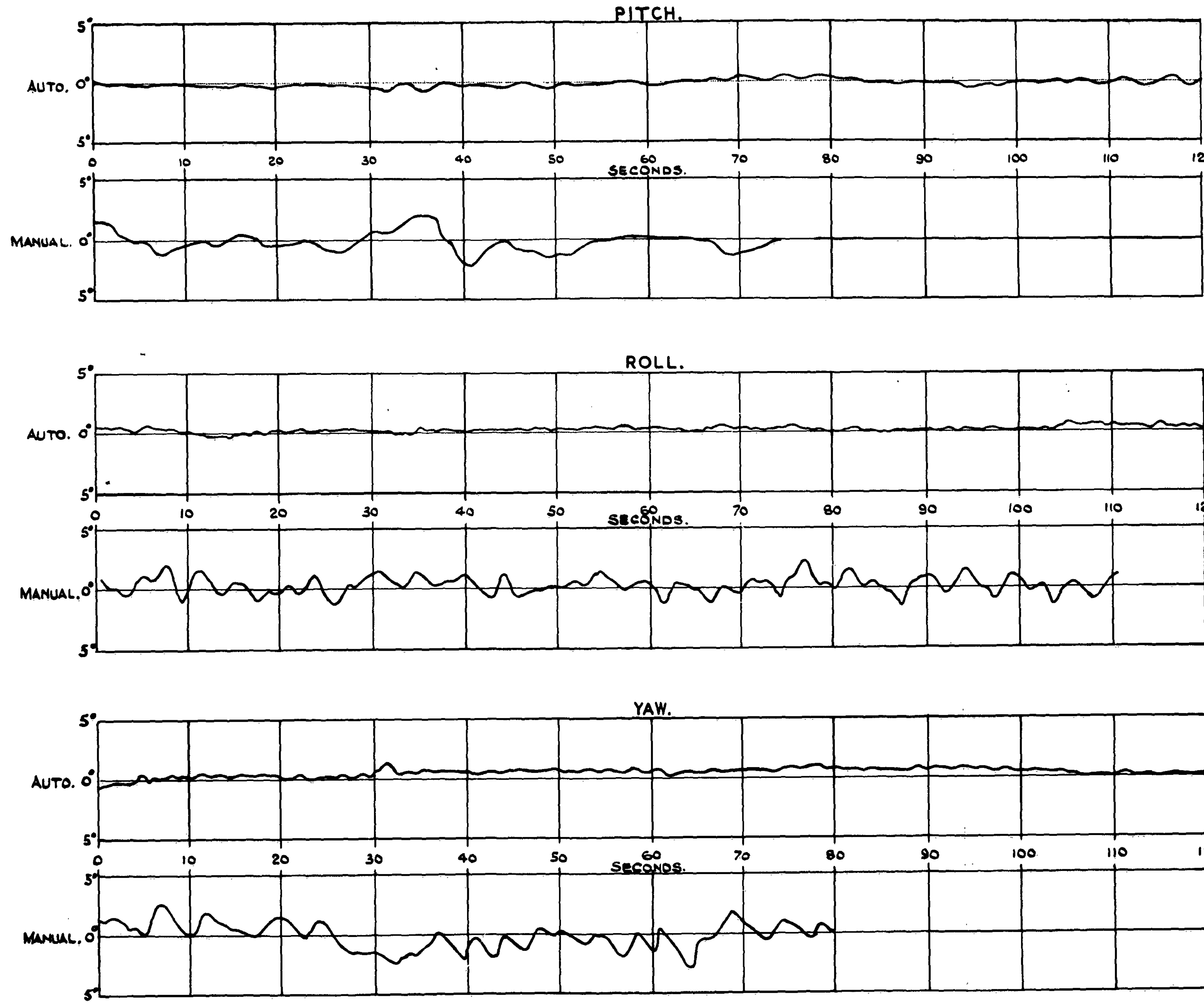


FIG. 6. WHIRLWIND H.A.R. MK. 22. (SIKORSKY. S.55) W.V.202. AUTOPILOT TYPE K.
COMPARISON OF AUTOMATIC & MANUAL CONTROL IN CONDITIONS OF INTERMITTENT MILD TURBULENCE.
CRUISING FLIGHT. — 60 KNOTS.

FIG.7. WHIRLWIND H.A.S. Mk.22

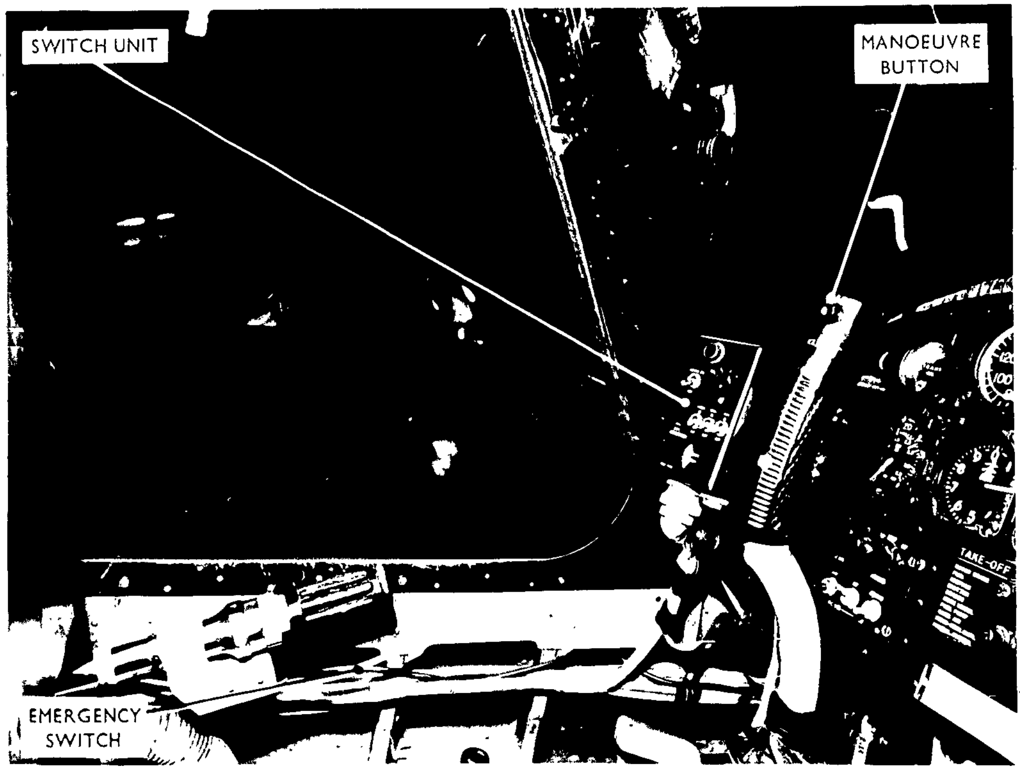


FIG.8 SECOND PILOT'S POSITION, PORT SIDE

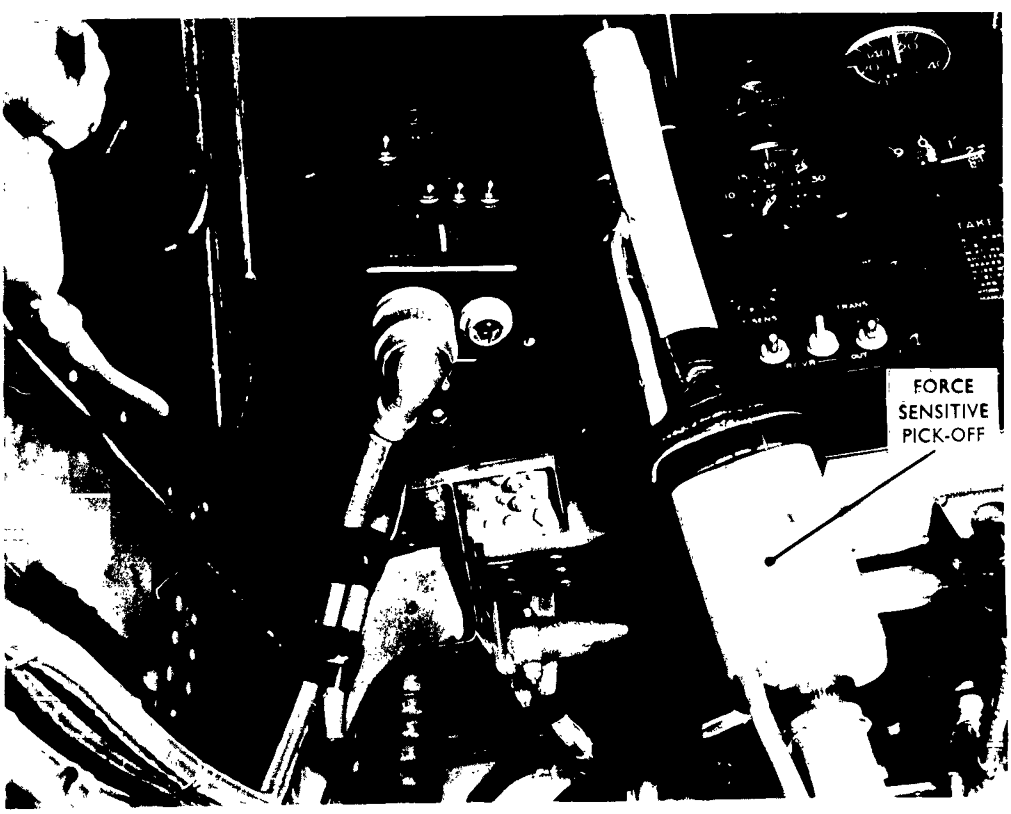


FIG.9. SWITCH UNIT AND CYCLIC STICK

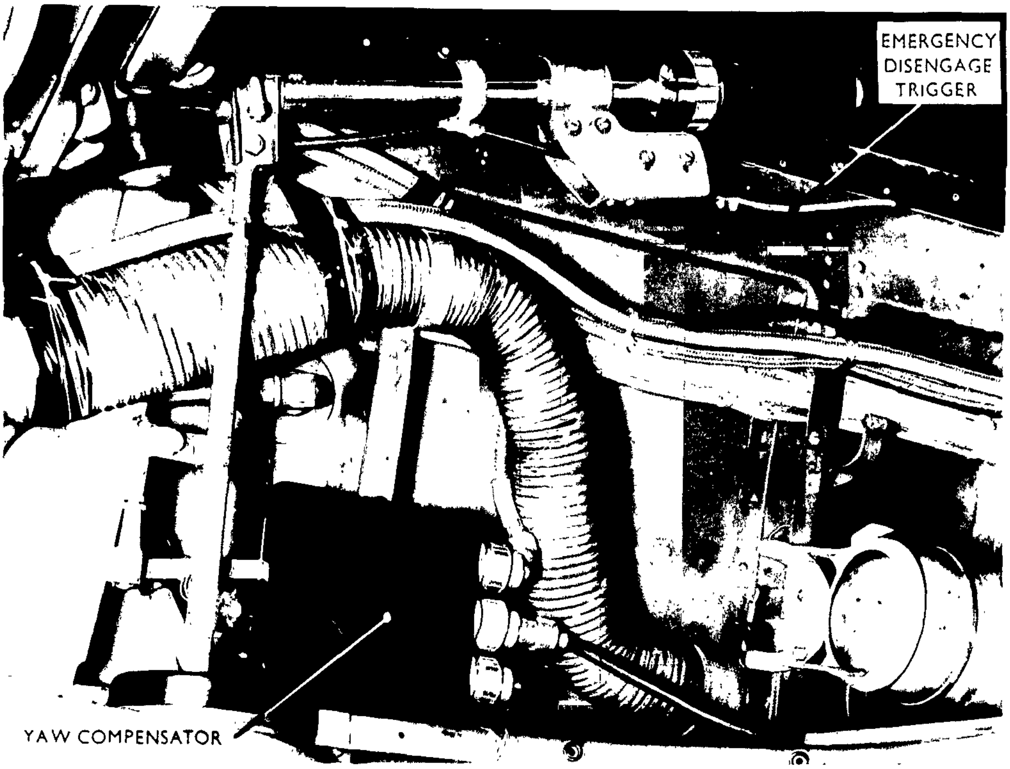


FIG.10. COLLECTIVE PITCH CONTROL, PORT SIDE

FIG.11. GYRO UNIT AND AMPLIFIER UNIT INSTALLATION

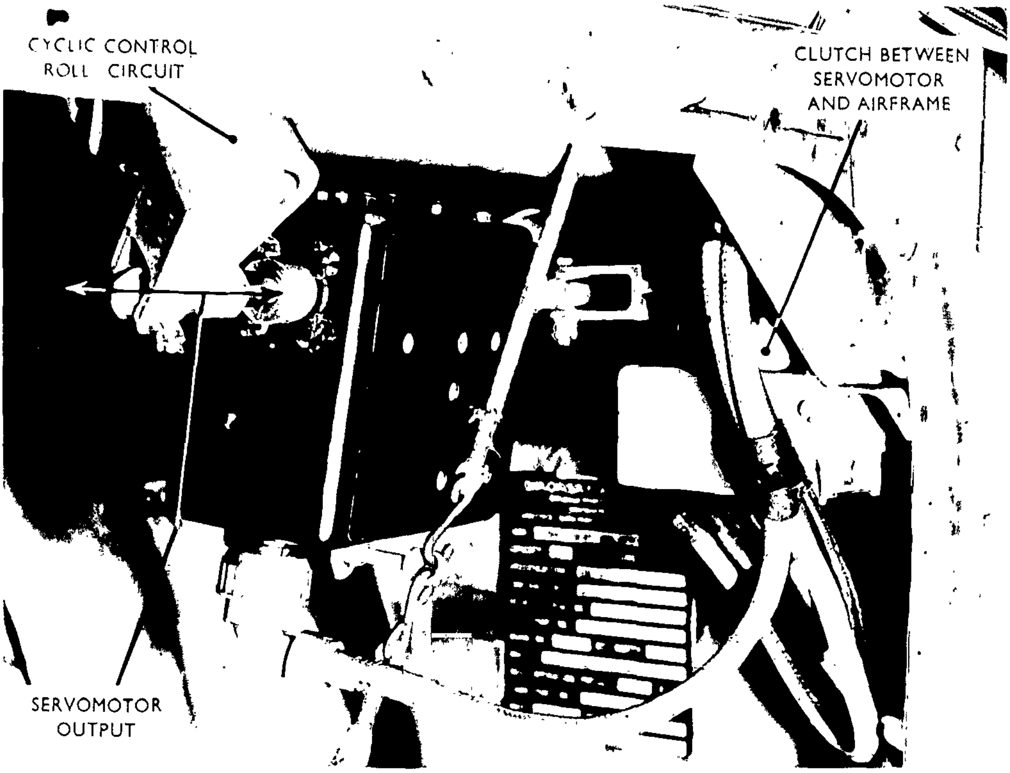


FIG. 12. ROLL SERVOMOTOR INSTALLATION

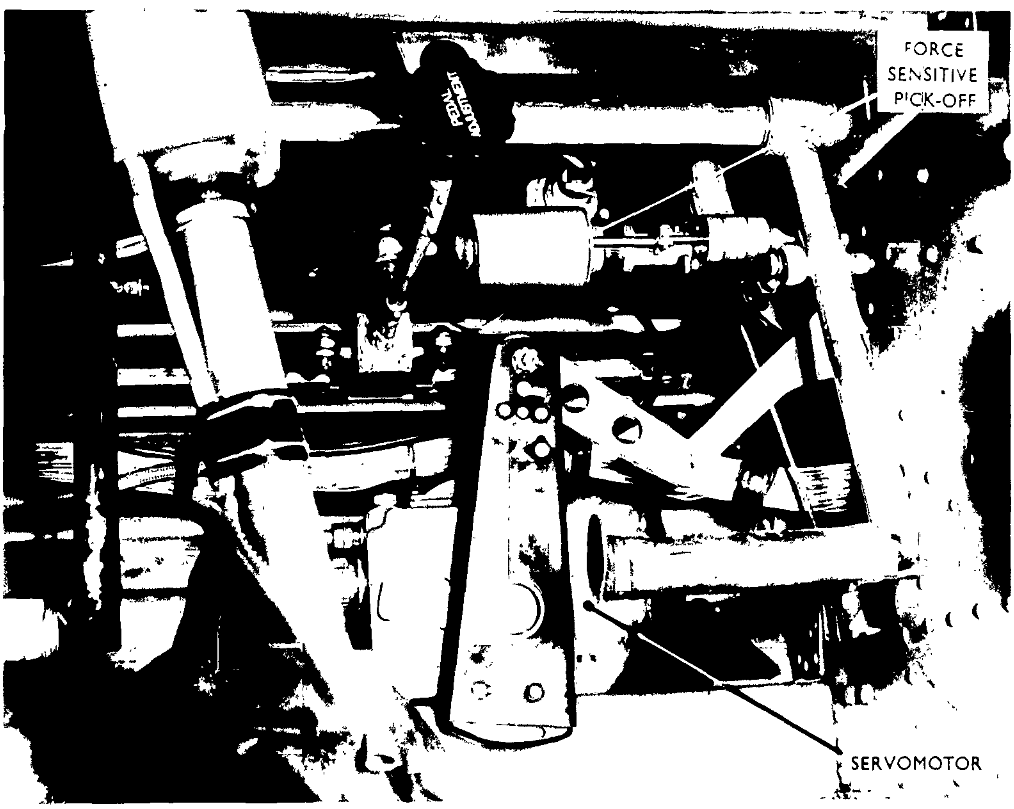


FIG 13. YAW SERVOMOTOR INSTALLATION

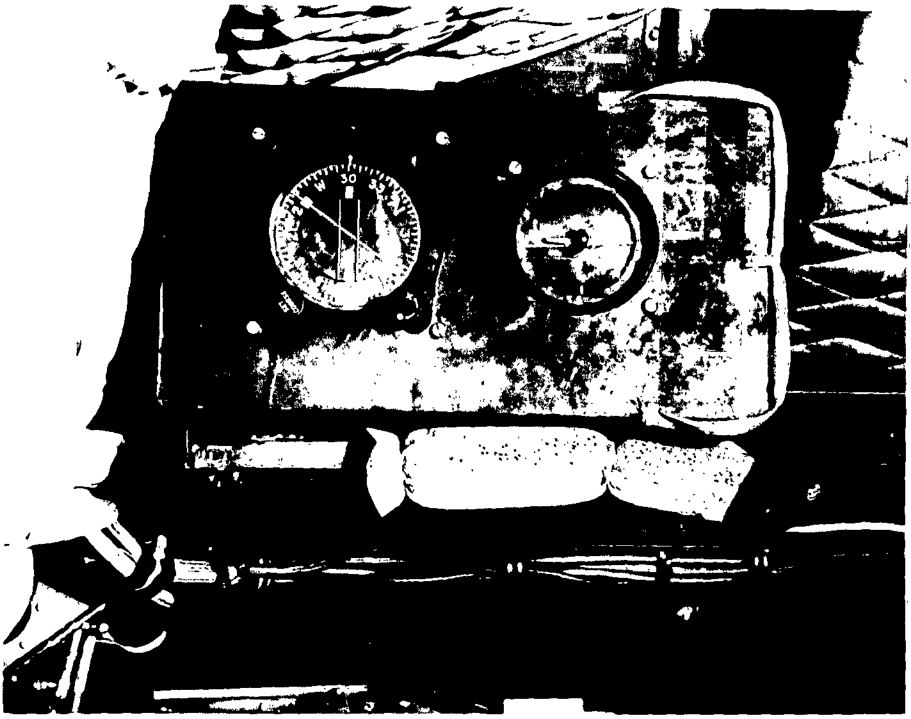


FIG.14 AUXILIARY INSTRUMENT PANEL

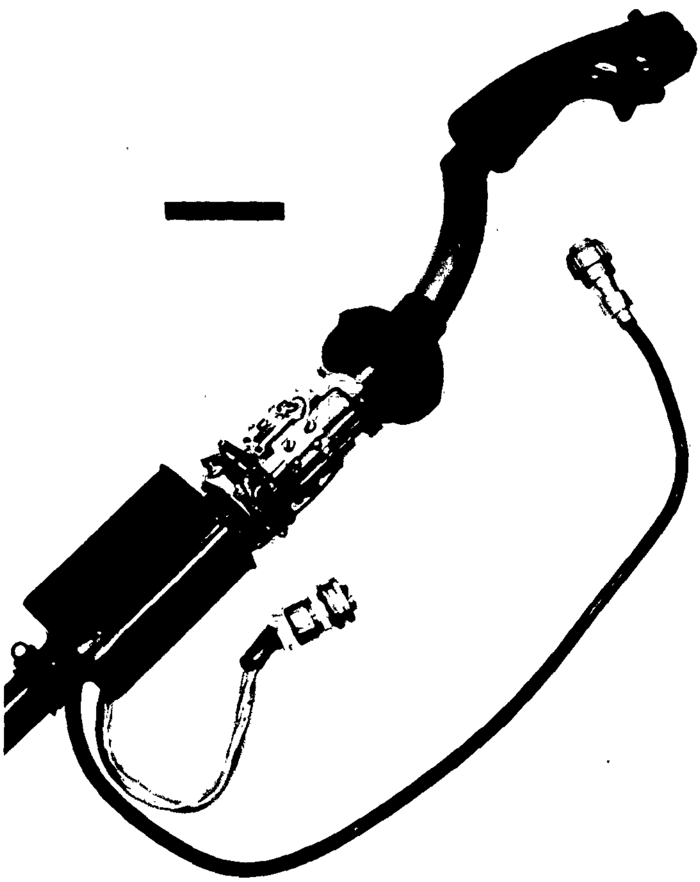


FIG 15 CYCLIC STICK

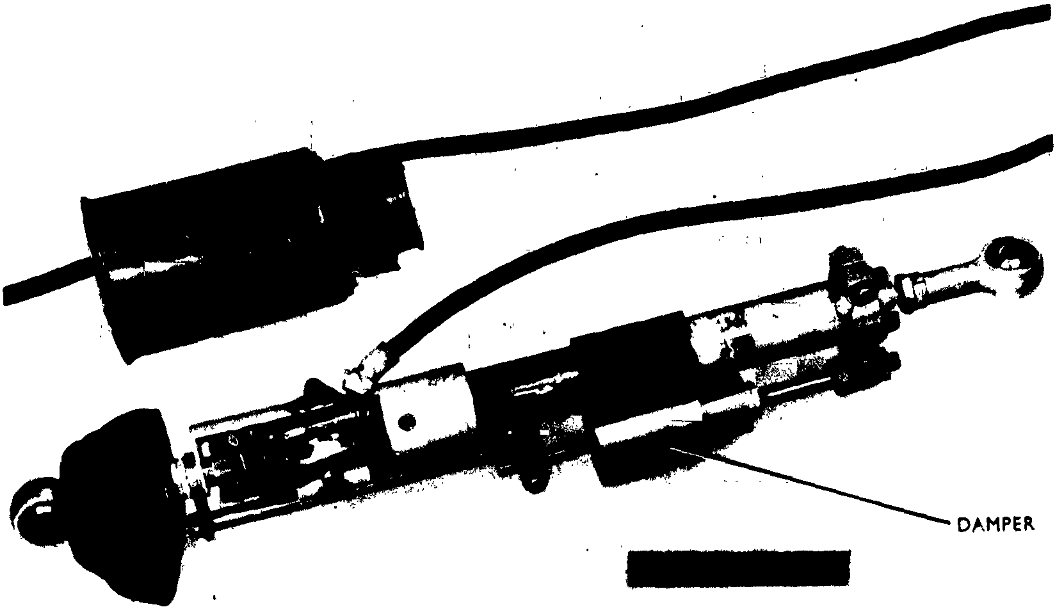


FIG. 16. YAW LINK

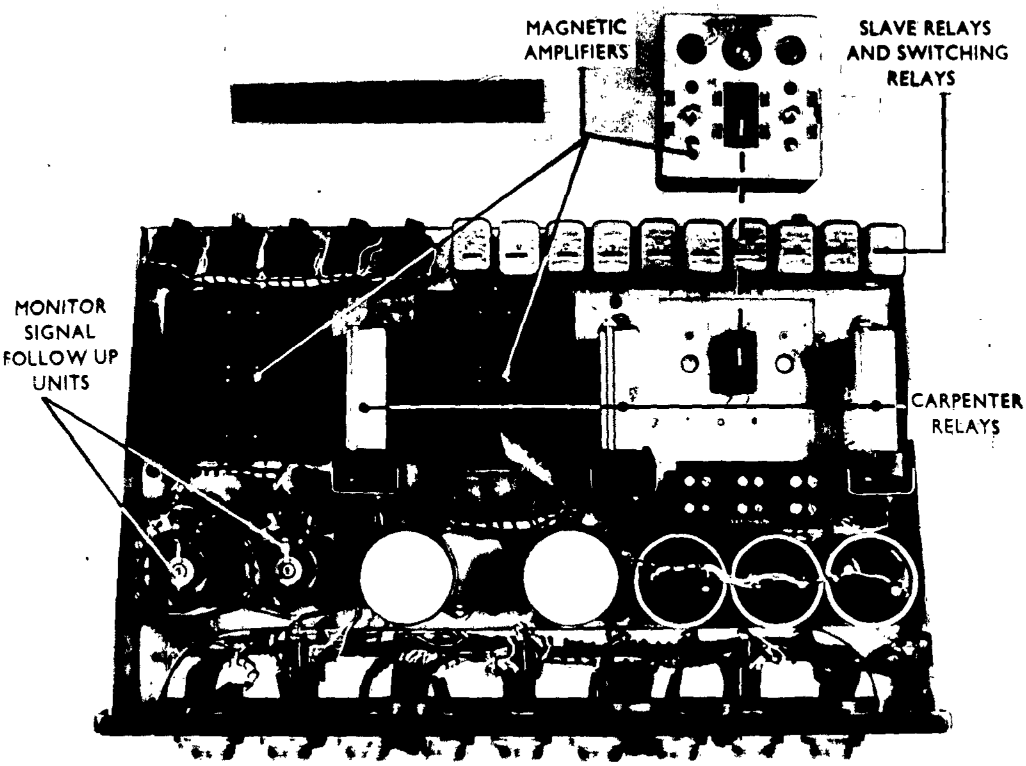


FIG 17 AMPLIFIER UNIT

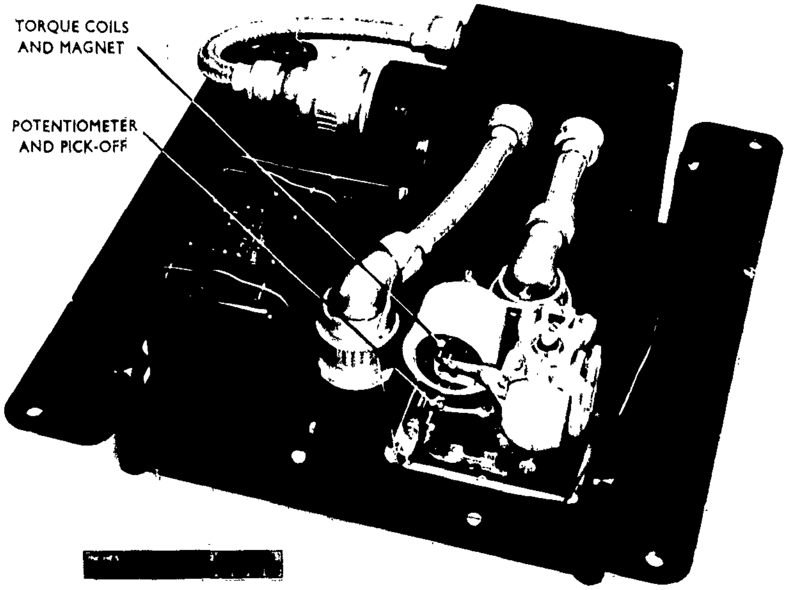


FIG. 18 GYRO UNIT

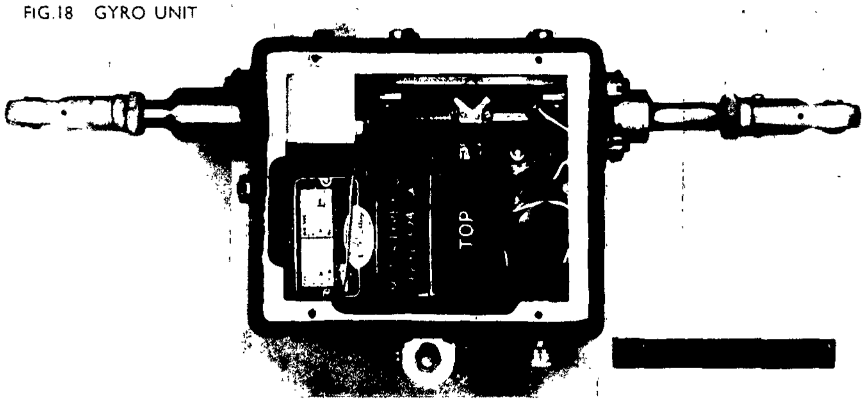


FIG. 19. PITCH AND ROLL SERVOMOTOR COVER REMOVED

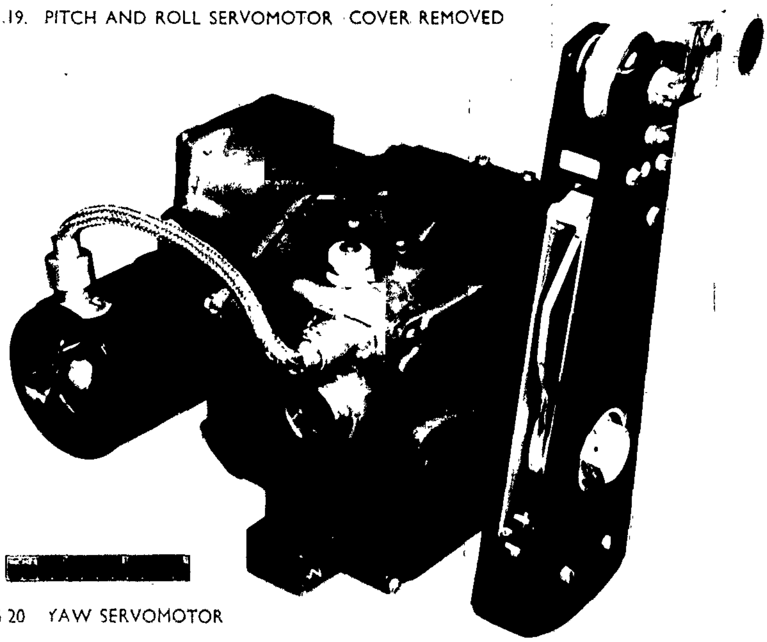


FIG 20 YAW SERVOMOTOR

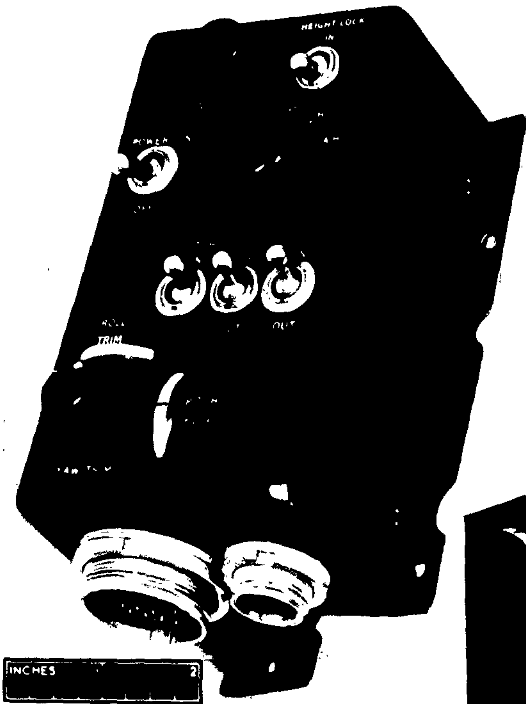


FIG 21 SWITCH UNIT

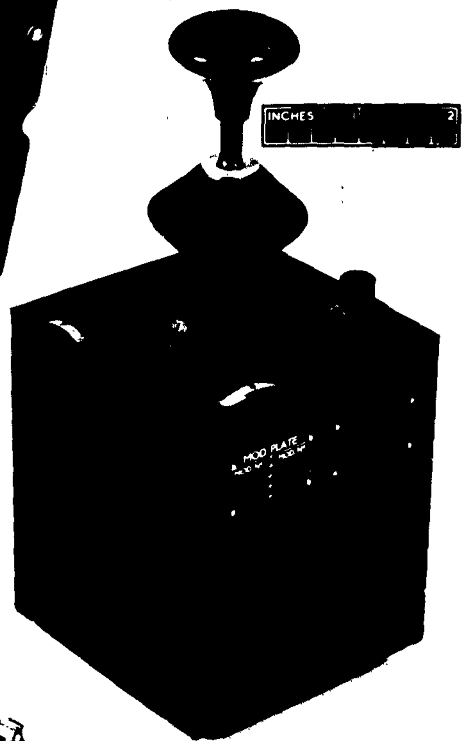


FIG.22 MINIATURE STICK
REMOTE CONTROLLER

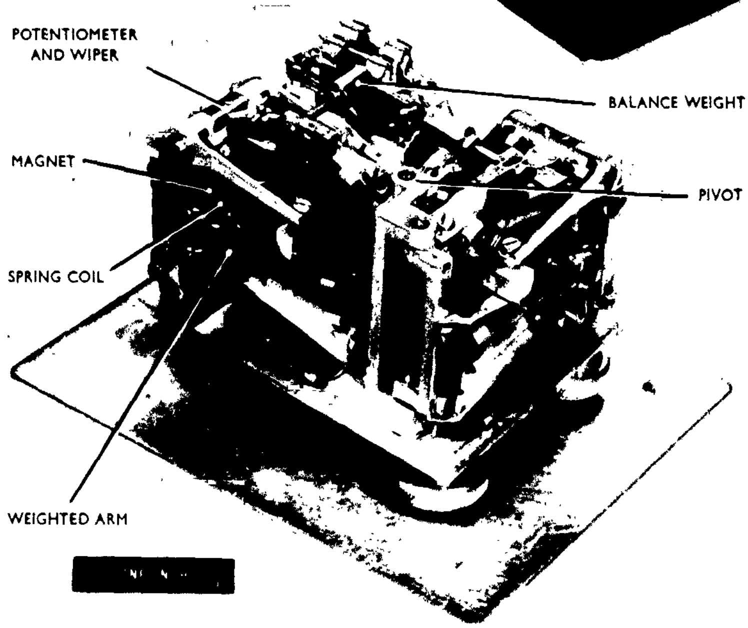


FIG.23. 2 AXIS ELECTRIC SPRING ACCELEROMETER
(NOTE:- ONLY THE COMPONENTS OF THE LATERAL ACCELEROMETER ARE INDICATED)

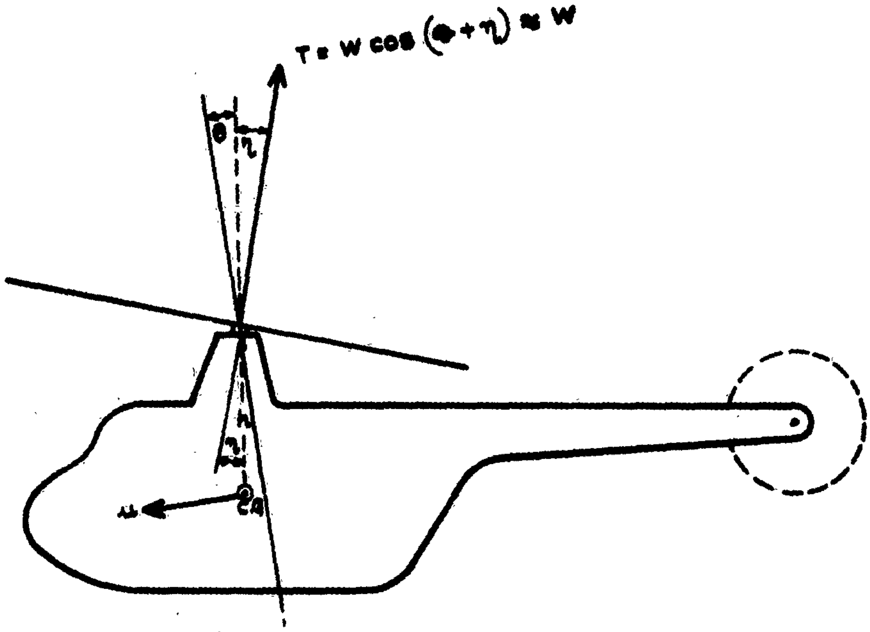


FIG. 24. SYMBOLS USED IN STABILITY APPENDIX. (PITCH.)

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AD#: AD091718

Date of Search: 5 August 2008

Record Summary: DSIR 23/24197
Title: Development of an automatic pilot for helicopters (RAE TN 1AP 1042)
Availability Open Document, Open Description, Normal Closure before FOI Act: 30 years
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